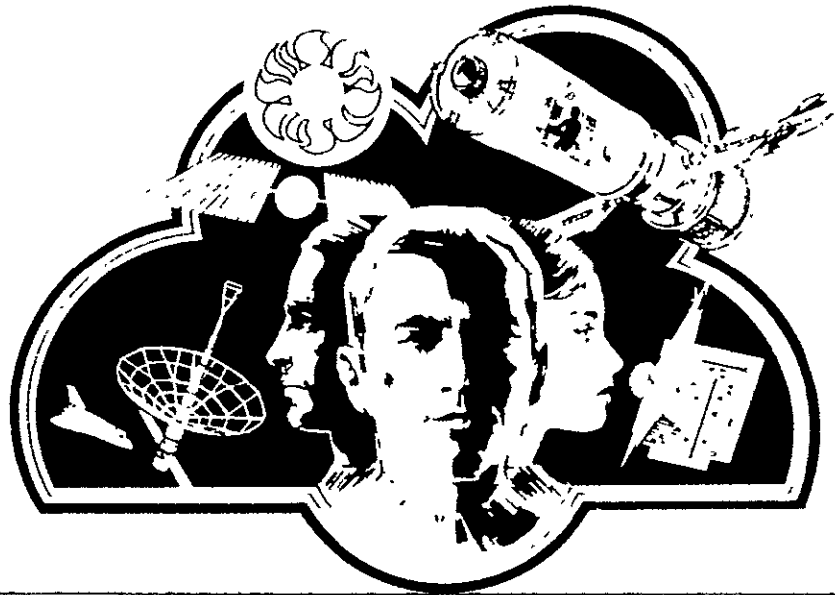




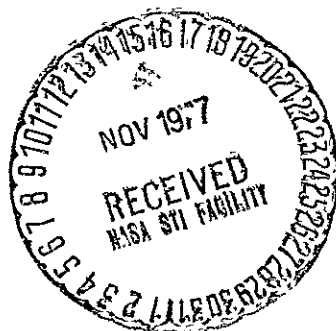
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## **SPACE STATION SYSTEMS ANALYSIS STUDY**

### **Part 3: Documentation**

### **Volume 7: SCB Alternate EPS Evaluation Task 10**

**MCDONNELL DOUGLAS ASTRONAUTICS COMPANY**



**MCDONNELL DOUGLAS**





# SPACE STATION SYSTEMS ANALYSIS STUDY

Part 3: Documentation

Volume 7: SCB Alternate Evaluation, Task 10

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## PREFACE

The Space Station Systems Analysis Study is a 15-month effort (April 1976 to June 1977) to identify cost-effective Space Station systems options for a manned space facility capable of orderly growth with regard to both function and orbit location. The study activity has been organized into three parts. Part 1 was a 5-month effort to review candidate objectives, define implementation requirements, and evaluate potential program options in low earth orbit and in geosynchronous orbit. Part 2 was also a 5-month effort to define and evaluate specific system options within the framework of the potential program options developed in Part 1.

Part 3, the last portion of this study, defines a series of program alternatives and refines associated system design concepts so that they satisfy the requirements of the low earth orbit program option in the most cost-effective manner. A separate Task 10 was added to the Part 3 portion of the study for evaluation of Space Construction Base (SCB) Power Sources. Alternative power sources were defined by various Government agencies for comparison with the baseline SCB solar array I battery system.

The final reporting of the Part 3 study activity consists of the following:

Volume 1, Executive Summary

Volume 2, Technical Report

Volume 3, Appendixes

Book 1, Supporting Data

Book 2, Supporting Data

Volume 4, Supporting Research and Technology Report

Volume 5, Cost and Schedules Data

Volume 6, Work Breakdown Structure and Dictionary

Volume 7, SCB Alternate EPS Evaluation, Task 10

A complete list of Parts 1 and 2 tables of contents are included for references in Volume 3, Book 2, Section 17.

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## Section 1 INTRODUCTION AND SUMMARY

This task was undertaken as part of the Space Station Systems Analysis Study (SSSAS), under joint sponsorship, of NASA and ERDA to update definitions of and evaluate potential alternatives to solar array battery systems for the Space Construction Base (SCB). The SCB is envisioned as a long-duration manned facility capable of supporting manufacturing and large-scale construction projects in space in addition to the more conventional R&D activities. Power levels up to 100 kWe average were baselined for the electrical power system, using data from earlier SSSAS activities.

Six alternatives to the solar array battery systems were specified for study in the contract statement of work these were: (1) solar concentrator/brayton, (2) solar concentrator/thermionic, (3) isotope brayton, (4) nuclear/brayton, (5) nuclear/thermoelectric, and (6) nuclear/thermionic.

Data packages containing extrapolations of past studies and current technology were supplied to MDAC by NASA/ERDA to serve as the starting point for this effort. In addition, persons from NASA and ERDA were designated as points of contact for each system.

Brief descriptions of the systems are given in the following text:

### SOLAR CONCENTRATOR/BRAYTON

The power system uses a 21-meter paraboloidal mirror to collect and focus solar energy in a heat receiver which transfers the energy to the brayton engine, which is a recuperated, closed-cycle, gas turbine. Lithium fluoride is incorporated in the receiver as a phase-change heat storage element that permits continuous power generation around the orbit.

In this system concept, one 60 kWe power module can be delivered in each Shuttle launch with two modules required to meet the combined growth and peak power requirement for the SCB. The concentrator and the heat storage elements are sized for the worst-case eclipse period in the baseline orbit.

#### SOLAR CONCENTRATOR/THERMIONIC

The thermionic power system also uses a solar concentrator to focus solar energy in a receiver and directly onto the thermionic diodes. A 27-meter concentrator is used to provide the 1,400 to 1,500 K diode temperatures needed and to accommodate the reradiation losses at these temperatures. Since the system does not incorporate a heat storage system, a separate energy storage element (batteries or regenerative fuel cells) is needed and the individual module output (115 kWe) is also sized to recharge the storage element during the sunlight periods.

At these temperatures, the pointing and surface accuracy requirements are more stringent than those of the solar brayton. Two power modules are used to meet the growth power needs of the SCB (100kWe).

#### ISOTOPE BRAYTON

This power system is under development by ERDA and can produce from 0.5 to 2.2 kWe per module. The system uses a small scale, brayton cycle, gas turbine engine similar to that proposed for the solar brayton system. Isotope fuel (Pu-238) capsules provide the system heat input, and the entire system is very compact.

#### NUCLEAR BRAYTON

The power system uses a high-temperature heat-pipe reactor design concept provided by the Los Alamos Scientific Laboratory (LASL) under an ERDA contract. The fast reactor proposes a uranium carbide fuel and transfers fission heat by 91 molybdenum-sodium heat pipes. The high-temperature source drives a gas turbine brayton engine at 1,273K. Waste heat from the cycle is rejected via a low-temperature heat-pipe radiator. Heat-pipe application is unique in power system designs and provides freedom from single-point failures.

A net cycle efficiency of 25 percent is anticipated. The system design will permit operation at either the initial or growth power levels by increasing the helium-xenon gas content in the brayton engine when increased power is needed. To permit this dual power capability, the engine, reactor shield, and radiator have been designed for the maximum power required. For power continuity and reliability reasons, a standby reactor is maintained at the SCB.

#### NUCLEAR THERMOELECTRIC

The nuclear thermoelectric power system uses the same heat-pipe reactor concept as described for the nuclear brayton system, but because of lower converter efficiencies requires a significantly higher reactor power level. Static thermoelectric converters interface directly, with the converter mounted around each heat pipe as it emerges from the reactor. The heat pipes and converters are arrayed as the surface of a conical frustrum. A complex arrangement of circumferential heat pipes is used around each thermoelectric converter and again between converters to assure multiple heat paths to the longitudinal radiator heat pipes. The entire assembly including the reactor presents significant manufacturing and assembly tasks.

#### NUCLEAR THERMIONIC

The nuclear thermionic power system also uses a LASL heat-pipe reactor concept. However, a uranium oxide ( $\text{UO}_2$ ) fuel is needed to attain the higher temperature levels (1,675K) for efficient thermionic operation. As a consequence of the fuel change, the thermionic reactor must be slightly larger and the size increase imposes some weight penalties. As with the thermoelectric power system, a complex interface is needed between the heat-pipe radiator and the cold junction of the diodes. Design detail is minimal in this area, and fabrication difficulties can be expected.

#### BASELINE POWER PLATFORM

The baseline power platform is assembled in orbit using space-manufactured composite beams and solar electric propulsion (SEP) solar cell and blanket technology. The active array size is 24 meters wide by 100.3 meters long.

Installation volume and surface area for the NiCd batteries, the power conditioning equipment, and waste heat radiators are provided by the fixture that is used during beam fabrication and array assembly. The rectangular array incorporates a one-degree-of-freedom ( $\pm 180^\circ$ ) gimbal at the interface between the array and the SCB. An average power level of 100 kWe is obtained with a maximum array output (sunlight) of 233 kWe at the beginning of life.

## EVALUATION OF SYSTEMS

During the first two months of the study, a top-level evaluation was made of the applicability and risk associated with these systems. By agreement with NASA/ERDA at the start of the contract, the isotope brayton power system was considered only for crew emergency power rather than for primary power due to its relatively high cost and the number of units required to produce the SCB power levels. Remaining power system alternatives were considered within solar and reactor heat source categories. The result was that the solar concentrator brayton and reactor (nuclear) brayton were selected for more detailed definition and evaluation. Section 3.4 of this report documents the power system assessment and selection, but summary comments are included here for convenience.

- Major elements of the solar brayton power system have been demonstrated for an 11 kWe size by operating one brayton engine for more than 30,000 hours at the system design temperature of 1,088K. Testing has also been done on the lithium fluoride heat storage elements. Scaling of the power elements to 60 kWe will be required. Fabrication, assembly, and alignment of the large-size concentrator remains to be demonstrated as does long-term compatibility of the heat storage-heat transfer elements.

Development of diodes for the solar thermionic power system has not been completed, and NASA projects low confidence that system efficiency can be met in the SCB time frame. All elements of the system require further design, analysis, and demonstration. For example, the design calls for stainless steel-mercury heat pipes to cool each diode at significantly higher temperatures than have been

demonstrated. The on-off thermal cycling of the converters in each orbit is also of concern relative to the system lifetime.

- The reactor brayton power system concept can readily meet SCB requirements with a design capable of operation at either the initial or growth power level. Development of the brayton cycle engine will require use of refractory materials to accommodate high system temperatures. The reactor design (using uranium carbide fuels) requires demonstration of the compatibility of interfacing existing technologies at new temperatures and power densities.
- Two converter materials have been proposed for the reactor thermoelectric system design. First is the silicon germanium thermoelectric material that has been used widely for isotope power systems in space and has demonstrated long lifetimes at lower temperatures and power levels. The second is a selenide thermoelectric material currently under ERDA development that promises higher system efficiency, but its technology readiness has not been demonstrated. With either converter material, it is not possible to meet growth power requirements without significant reactor and power system redesign. The reactor design comments given under the reactor brayton system are also applicable here.
- The efficiencies projected for the reactor thermionic power system design have not yet been attained in thermionic diode development. Demonstration of interfacing technology designs is also lacking, and the paucity of design detail shows this system to be the least mature of those considered.

During the final four months of the study, the two alternative power systems were analyzed relative to the evolving SCB configurations to provide performance and cost data for comparison with the baseline SCB power platform. In addition, an alternate photovoltaic system design using deployable SEP arrays was developed. The alternate deployable array system uses low-cost solar cells (\$5/watt) and a regenerative fuel cell energy storage system.

Application of these advanced technology developments-one at a time-to the baseline power platform provided four possible solar photovoltaic system designs for comparison with the solar and reactor brayton alternatives.

#### SUMMARY COMPARISONS

The two alternative power systems were compared to the four photovoltaic designs in terms of the seven criteria provided for system assessment. The criteria included cost, safety, performance, design complexity, operations complexity, development constraints, and program flexibility. The most significant conclusions in the major programmatic areas of comparison are discussed in the following paragraphs.

##### Cost

Ten-year program costs for the baseline power platform, the solar brayton, and the reactor brayton power systems are compared in Figure 1-1.

Historical data and projected risk for each were assessed to establish cost uncertainties as shown by the "probable upper bound cost" figures. Cost factors (see Section 7.2 for rationale) as applied to the DDT&E and production costs were 25 percent for the power platform, 50 percent for the solar brayton, and 100 percent for the reactor brayton. A cost advantage of \$52 million is shown for the solar brayton relative to the power platform and \$242 million relative to the reactor brayton.

Alternate photovoltaic systems were examined, and Figure 1-2 provides cost data for four such systems. Both the power platform and advance solar array systems have approximately the same cost; however, application of low-cost solar cells or regenerative fuel cells to the power platform indicate costs equivalent to those of the solar brayton. Not included is the effect of longer lifetime NiCd or NiH<sub>2</sub> batteries which give promise of even lower photovoltaic system costs. System 10-year cost comparisons show relatively minor differences.



|                                       | POWER PLATFORM* | SOLAR<br>BRAYTON | REACTOR<br>BRAYTON |
|---------------------------------------|-----------------|------------------|--------------------|
| DDT&E                                 |                 |                  |                    |
| SOLAR ARRAY AND SUPPORT STRUCTURE     | 36              | —                | —                  |
| ENERGY STORAGE AND POWER CONDITIONING | 18              | —                | —                  |
| STRUCTURE/MECH/MISC                   | 19              | —                | —                  |
| POWER MODULE                          | —               | 103              | 179                |
| INTEGRATION PACKAGE                   | —               | 14               | 21                 |
| SUBTOTAL                              | 73              | 117              | 200                |
| PRODUCTION                            |                 |                  |                    |
| SOLAR ARRAY AND SUPPORT STRUCTURE     | 70              | —                | —                  |
| ENERGY STORAGE AND POWER CONDITIONING | 32              | —                | —                  |
| STRUCTURE/MECH/MISC                   | 2               | —                | —                  |
| POWER MODULES (\$)                    | —               | 8                | 18                 |
| INTEGRATION PACKAGE                   | —               | 2                | 3                  |
| SUBTOTAL                              | 104             | 10               | 21                 |
| OPERATIONS (10 YEARS)                 |                 |                  |                    |
| INITIAL LAUNCH (\$20M/FLIGHT)         | 24              | 40               | 40                 |
| SUPPORT LAUNCHES                      | 30              | 42               | 42                 |
| SPARES AND REPLACEMENT HARDWARE       | 33              | 5                | 5                  |
| RCS PROPELLANT COST                   | 40              | 18               | 9                  |
| SUBTOTAL                              | 127             | 105              | 96                 |
| TOTAL                                 | 304             | 232              | 317                |
| PROBABLE UPPER BOUND**                | 348             | 296              | 538                |

\*SEP TECHNOLOGY

\*\*BASED ON HISTORICAL AND RISK ASSESSMENT

Figure 1-1. Ten-Year Power/Program Costs (Millions of 1978 Dollars)

### Safety

Electrical power system safety does not appear to be a major problem even though the reactor system will require significant analytical effort to ensure safe disposal. As a consequence, higher safety costs can be expected for the reactor system.

### Performance

For both initial and growth power levels, all the candidate systems can meet SCB requirements with about the same 10-year system reliability. The solar brayton system has the best weight growth margin, followed closely by the reactor brayton system. However, any unexpected weight growth for the photovoltaic systems can be accommodated with only a nominal change in launch costs.

|                                       | SOLAR PHOTOVOLTAIC                       |                                            |                                    |                                |
|---------------------------------------|------------------------------------------|--------------------------------------------|------------------------------------|--------------------------------|
|                                       | POWER PLATFORM                           |                                            | SEPS SOLAR CELLS<br>AND FUEL CELLS | ADVANCED<br>DEPLOYABLE ARRAY** |
|                                       | SEPS SOLAR CELLS *<br>AND NiCd BATTERIES | LOW-COST SOLAR CELLS<br>AND NiCd BATTERIES |                                    |                                |
| DDT&E                                 |                                          |                                            |                                    |                                |
| SOLAR ARRAY AND SUPPORT STRUCTURE     | 36                                       | 50                                         | 36                                 | 42                             |
| ENERGY STORAGE AND POWER CONDITIONING | 18                                       | 18                                         | 40                                 | 40                             |
| STRUCTURE/MECH/MISC                   | 19                                       | 19                                         | 19                                 | 55                             |
| SUBTOTAL                              | 73                                       | 85                                         | 95                                 | 137                            |
| PRODUCTION                            |                                          |                                            |                                    |                                |
| SOLAR ARRAY AND SUPPORT STRUCTURE     | 70                                       | 27                                         | 70                                 | 34                             |
| ENERGY STORAGE AND POWER CONDITIONING | 32                                       | 32                                         | 20                                 | 20                             |
| STRUCTURE/MECH/MISC                   | 2                                        | 2                                          | 2                                  | 6                              |
| SUBTOTAL                              | 104                                      | 61                                         | 92                                 | 60                             |
| SUBTOTAL, DDT&E AND PRODUCTION        | 177                                      | 148                                        | 187                                | 197                            |
| OPERATIONS (10 YEARS)                 |                                          |                                            |                                    |                                |
| INITIAL LAUNCH (\$20M/FLIGHT)         | 24                                       | 24                                         | 24                                 | 40                             |
| SUPPORT LAUNCHES                      | 30                                       | 30                                         | 2                                  | 2                              |
| SPARES AND REPLACEMENT HARDWARE       | 33                                       | 33                                         | 16                                 | 16                             |
| RCS PROPELLANT COST                   | 40                                       | 40                                         | 40                                 | 38                             |
| SUBTOTAL                              | 127                                      | 127                                        | 82                                 | 96                             |
| TOTAL                                 | 304                                      | 275                                        | 269                                | 293                            |
| PROBABLE UPPER BOUND                  | 348                                      | 312                                        | 316                                | 342                            |

NOTE: SHUTTLE-ERECTED FREE-FLIGHT POWER PLATFORM REQUIRES ADDITIONAL \$37 MILLION DDT&E AND \$18 MILLION PRODUCTION COST  
COMPOSITE BEAM FABRICATION AND ASSEMBLY COST DELETED FROM POWER PLATFORM COSTS: EQUIPMENT  
USED ON OTHER CONSTRUCTION.

\*BASELINE

\*\*NO SPACE FABRICATION - FUEL CELL ENERGY STORAGE - 12% - \$5/W SOLAR CELLS

Figure 1-2. Ten-Year Solar Array Program Costs (Millions of 1978 Dollars)

### Design Complexity

While the solar and reactor brayton power systems do not require separate eclipse or peak power subsystem elements, each design is new and not as well defined as the solar photovoltaic systems. The latter have been used for the majority of long-duration spacecraft and satellites; consequently, the integration and operation of solar cells and batteries are well understood. By virtue of experience, solar photovoltaic systems are judged to be readily acceptable power systems.

### Operations Complexity

Analyses during this study have shown that the reactor power system is least obtrusive on the SCB operational volumes (i.e., volumes either occupied or swept). Further, the radiation analysis has shown that the reactor can be integrated readily with astronaut extravehicular activity (EVA) and in-module exposure. The weight penalty for accommodation of the radiation exposure is minimal. While the solar array and solar brayton power systems occupy significant volume, their impact is primarily in obstructing viewing or impeding freedom of movement.

All the candidate power systems constrain SCB orientations to ensure either solar illumination or reasonable attitude control requirements. Thus, specific inertial pointing requirements probably should be accommodated by those mission hardware elements that have inertial viewing needs.

### Development Constraints

Clearly, past development and applications have established the photovoltaic systems as the only ones that can immediately allow design definition with minimum risk. Unless funding is provided for alternative systems to accomplish the needed supporting research and technology (SR&T), this situation will remain unchanged. Neither the solar nor the reactor brayton power systems can be ready for the mid-1985 IOC for the SCB without development starts in Fiscal 1978, well in advance of normal power system selection. Neither system has attained technology readiness and could not meet the IOC date with a design go-ahead in Fiscal 1980.

### Program Flexibility

Consideration of the candidate power systems for different mission and space-craft requirements has been based on high-level requirements. Briefly, the reactor power systems have the greatest power growth capability. In terms of total weight to orbit, three of the systems show nearly equal requirements for weight and numbers of launches (Table 7-4). Only the baseline power platform shows significant weight difference, and, as noted earlier, the application of new technology for low-cost solar cells and regenerative fuel cells eliminates this weight differential.

All have nearly equivalent attainment of credible low-earth orbit altitudes and inclinations. At geosynchronous orbit, the photovoltaic systems would gain the most by virtue of near-continuous illumination. Gains also would accrue for the solar brayton.

Table 1-1 is a tabular comparison the the power systems, and highlights the fact that only the solar photovoltaic systems can meet all the criteria established for SCB power system evaluation. The solar cell system rates "good" or better in all areas except in the operations complexity region where the large array size constrains viewing and freedom of access. For the defined SCB activities, the net viewing limitations do not cause major constraints and the station design accommodates the access constraints.

Each of the power systems receives above average ratings in one or more areas and only in the area of technology readiness (development constraints) can a clear selection be made. A solar photovoltaic power system with either state-of-the-art batteries or one of the advanced technology storage systems is recommended.

| POWER SYSTEM       | COST         | SAFETY       | PERFORMANCE | DESIGN COMPLEXITY | OPERATIONS COMPLEXITY | DEVELOPMENT CONSTRAINTS | PROGRAM FLEXIBILITY |
|--------------------|--------------|--------------|-------------|-------------------|-----------------------|-------------------------|---------------------|
| SOLAR PHOTOVOLTAIC | GOOD         | GOOD         | GOOD        | VERY GOOD         | FAIR                  | BEST                    | VERY GOOD           |
| SOLAR BRAYTON      | VERY GOOD    | GOOD         | GOOD        | GOOD              | GOOD                  | POOR                    | GOOD                |
| REACTOR BRAYTON    | FAIR TO POOR | GOOD TO FAIR | GOOD        | GOOD              | VERY GOOD             | POOR                    | VERY GOOD           |

ORDER OF RATINGS: ● BEST ● VERY GOOD ● GOOD ● FAIR ● POOR

Table 1-1. Power System Comparison

### SUMMARY OBSERVATIONS

Since the reality of a mid-1985 IOC for an SCB is in serious question at this time, elimination of either the reactor or solar brayton systems for the reason of technology readiness is not recommended. The following observations are offered for NASA/ERDA consideration in light of probable delays for the SCB.

- The bulk of SCB requirements identified to date can be satisfied by power levels of 40 to 60kWe, although an early solar power satellite antenna test program would require higher power levels. Inasmuch as the timing of such a test program is not known, further efforts to define an SCB power system should concentrate on providing extended operations at the 40- to 60-kWe power level while ensuring an easy growth path to the 100- to 120kWe level (i.e., no significant system or module redesign should be required to achieve growth).
- Design definition is needed of a solar array power system, whether deployable or constructed in space, which takes maximum advantage of near-term advances in technology. Within this context, low-cost solar cell blanket design and fabrication, design and operation of either a regenerative fuel-cell energy storage system or extended-life NiCd or NiH<sub>2</sub> batteries, and the application of solidstate electronics to the power system control and distribution functions should be investigated.

- Design definition and costing of a solar brayton power system is needed concentrating on the problems of concentrator/receiver fabrication, assembly, alignment, and maintenance. In conjunction with this design effort, SR&T should be encouraged in the same areas along with basic material compatibility studies and tests if necessary.
- A reactor brayton SR&T program that develops the key technology areas (e.g., fuel heat-pipe performance, heat-source heat-exchanger interfaces with the brayton engine, brayton engine design and layout, the heat rejection system including brayton engine and radiator interfaces, and disposal-safety requirements) should be encouraged. The reactor brayton power system has shown discrete advantages in terms of operational freedom and power growth capability. However, the system is the least well defined and the SR&T needs are the largest. ERDA should be encouraged to pursue work on its concept, particularly in key technology areas. It is recommended that NASA monitor the ongoing AF/ERDA work and maintain cognizance of ERDA progress in the key technology design and demonstration areas. A more detailed application study would depend on the availability of more adequate technology and system definition from ERDA.

## Section 2

### REQUIREMENTS EVOLUTION AND EFFECT

An initial set of EPS requirements was derived largely from the SSSAS Design Guidelines and Criteria Document, JSC-11867, dated January 1977, and from the mission power profile (Figure 2-1). Derivation of these requirements, summarized in Table 2-1, was also influenced by data from the ongoing SSSAS and past NAR and MDAC Phase B Space Station studies, and by appropriate engineering judgement.

The initial EPS requirements document, which was issued in February 1977, was used in the early portion of this study subsequent to the Part 3 Engineering Review Board meeting (April 14, 1977). These requirements were then updated, and are included as Appendix A of this document. The effect of these changes will be discussed later in this section.

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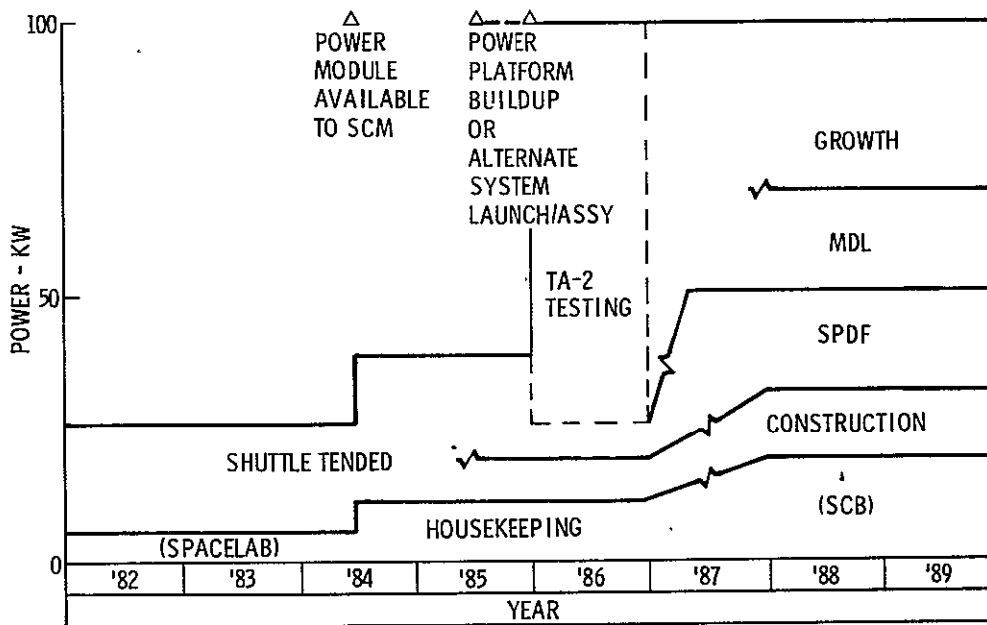


Figure 2-1. SCB Power Profile

Table 2-1

SSSAS REQUIREMENTS, JSC-11867, JANUARY 1977 (MODIFIED)

- ORBIT 370 TO 650 KM, 28.5 TO 90° INCLINATION, BASLINE 400 KM, 28.5° INCLINATION
- POWER 50 KWe 115/230 VAC, 3Φ, 400 AT SCB, GROWTH 100 KWe  
120% PEAK - 1 HOUR - 3 TIMES A DAY, 400+ KW PEAK ~ 15 MIN/ORBIT  
50% MINIMUM PARTIAL POWER DURING MAINTENANCE/REPAIR
- GOAL - NO ORIENTATION RESTRICTIONS
- LIFETIME - 10 YEARS WITH MAINTENANCE OR REPLACEMENT
- IOC - MID 1985 - USE 1980 PROVEN TECHNOLOGY
- STANDBY CAPABILITY - APPROXIMATELY 4 MONTHS
- SHUTTLE LAUNCH  
80% OF PAYLOAD CAPABILITY: 51K TO 400 KM, 28.5° INCLINATION  
PACKAGE SIZE 14.5'D X 52'MAX  
SURVIVE LAUNCH AND ABORT LOADS AND MEET CG CONSTRAINTS
- FAIL SAFE/FAIL OPERATIONS
- SAFETY - NO DANGER TO SCB OR CREW  
- PROVIDE FOR CHECKOUT AND MONITORING/DISPOSAL
- EXTENSIVE EVA AVAILABLE IF REQUIRED
- RADIATION ALLOCATION - 12.5 REM/3MONTHS

Of the requirements listed, the need initially for 50 kWe of power growing over a period of years to 100 kWe, and the need for 50% power during periods of maintenance and repair, provide a strong incentive to consider a growth system consisting of at least two modules, one of which could be used as the initial system. In addition, the 10-year-life goal indicates a need for a system having both a high level of redundancy and a reasonable level of maintainability.

The requirement to assure crew survival for 180 hours in event of a major catastrophe could be satisfied either by providing sufficient power to operate minimal station life support and thermal control systems or by supplying an independent self-contained emergency survival system.

The first approach requires a storage system capable of supplying approximately 5 kW of power per hour for the 180-hour period, or a total of 900 kW-hr. Figure 2-2 shows the weight penalty associated with addition of such an energy storage system. Of the candidates shown, fuel cells with gaseous storage are



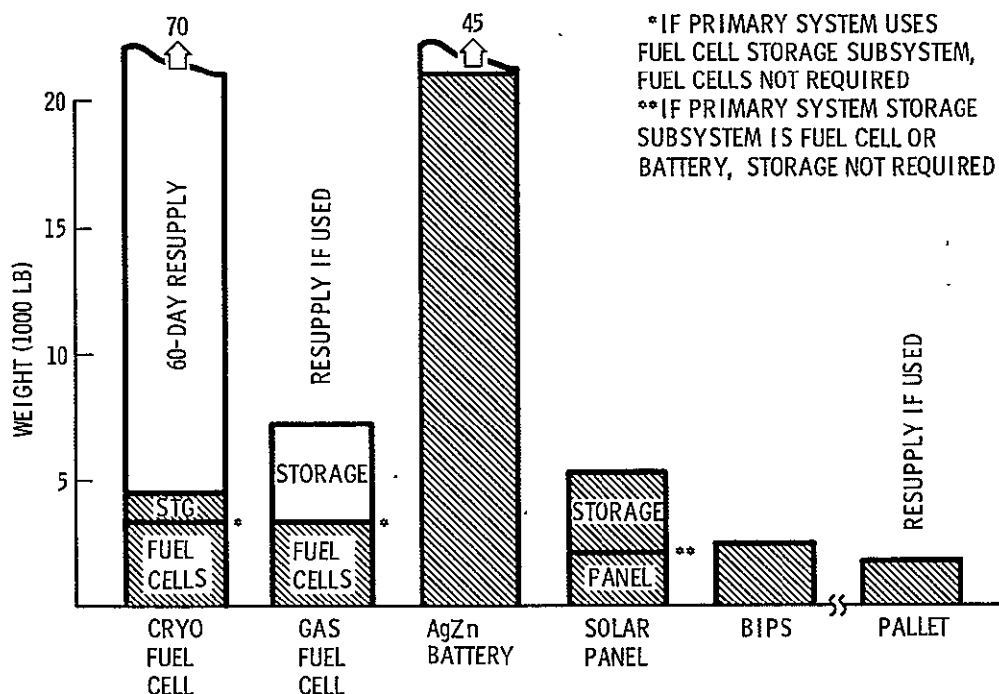


Figure 2-2. Emergency Systems 5 kW - 180 Hours

particularly attractive because they require a total weight of 8000 pounds or less depending on whether the primary EPS uses fuel cell storage. An additional solar panel system would also be attractive if the SCB configuration could accept installation of a rather large emergency system (about the size of the Skylab installation). The brayton isotope power system (BIPS), which currently is rated at a maximum of 2.2 kW, would require at least two units which would be inordinately expensive (\$3000 to \$5000 per We, excluding DDT&E—\$13.2 to 22M for 2 units).

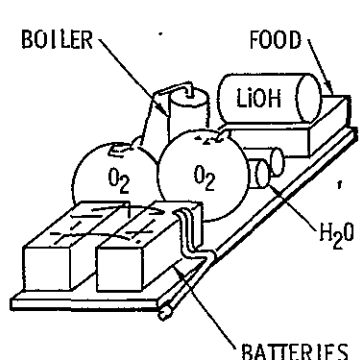
If the EPS is modularized sufficiently to provide assurance of 5 kW continuous power under all but catastrophic conditions, the use of an independent life support pallet (see Figure 2-2) becomes attractive. Figure 2-3 shows such a pallet conceived for SCB. This system has been baselined for the SCB thus removing the emergency requirement from the EPS system.

At the May 14, 1977 ERB meeting, early antenna tests for MDAC's SPS microwave power transmission system (MPTS) were selected as a prime

## CANDIDATE SCB LOCATIONS

- CONSTRUCTION SUPPORT MODULE
- LOGISTICS MODULE

CHARACTERISTICS: 7-MAN PALLET 180 HOURS



| PROVISION               | REQUIREMENT                      | WEIGHT<br>KG (LB) |
|-------------------------|----------------------------------|-------------------|
| OXYGEN                  | METABOLIC O <sub>2</sub>         | 103 (227)         |
| WATER                   | CREW INTAKE PLUS COOLING         | 282 (621)         |
| FOOD*                   | 2700K CAL DIET                   | 59 (131)          |
| LiOH                    | MAINTAIN 0.102 kN/M <sup>2</sup> | 83 (183)          |
| BATTERIES               | 4200 WATT-HR                     | 99 (219)          |
| WATER BOILER            | 3200 WATT (1600 BTU/HR) COOLING  | 14 (30)           |
| MISSCELLANEOUS SUPPLIES |                                  | 9 (20)            |
| PALLET/PACKAGING        |                                  | 49 (109)          |
| TOTAL                   |                                  | 698 (1540)        |

\*MAY NOT BE REQUIRED IN LOGISTICS MODULE

Figure 2-3. Emergency Pallet Concept

activity. This added a peak power requirement of approximately 450 kW for approximately 15 minutes several times a day (preferably once per orbit), a significant energy need compared with the initial baseline system output. In addition, an average initial power capability of 40 kWe (rather than 50) was selected with growth to 100 kWe still required.

Several alternative approaches to handling these new requirements were investigated. It was finally decided to use the growth EPS with added energy storage capability in order to handle these requirements. Upon completion of MPTS testing, the EPS could be operated at initial power capability for an interim period until the growth capability was again needed. The cost of supplying the growth capability early in the SCB program, while not desirable, is necessary if MPTS testing is to be an early objective of the SCB program.

The following paragraphs summarize the major characteristics of the four systems studied (two solar photovoltaic, the solar concentrator, and the

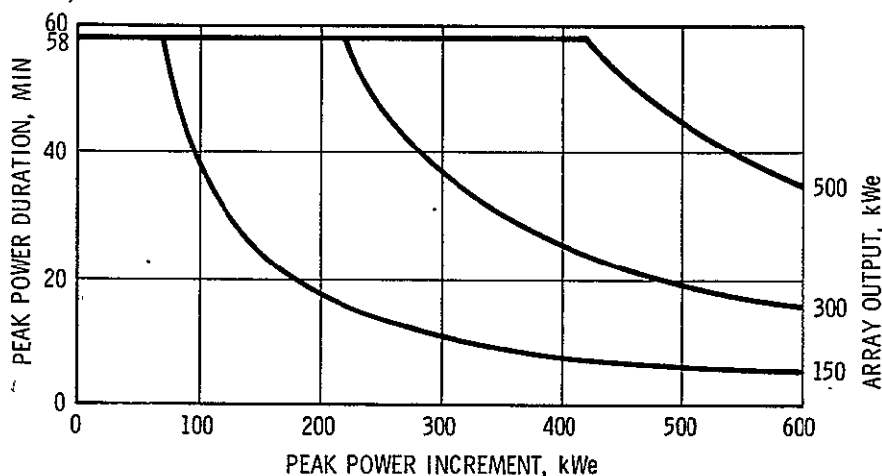
reactor). Rationale for selecting the number and size of engines, the energy storage systems, etc., is presented in subsequent sections of this report.

## 2.1 SOLAR PHOTOVOLTAIC SYSTEM

The baseline solar array system consists of a 250-kW array segmented into four electrically independent modules which, with an adequate battery storage system, can support the 100-kWe growth system requirement. For initial operations, batteries are supplied to support an average 40-kWe load; this requires full use of approximately 100 kW of the array power. The MPTS test antenna can be supplied with additional batteries which, when used in conjunction with the balance of the array, can supply the required peak power. Sample peak power performance curves are presented in Figure 2-4.

When MPTS testing has been completed and the SCB requires more than 40 kWe of power, batteries are added to the SCB as required until the 100-kWe average power level is achieved.

CR 75



- NOTES: 1. ONE CYCLE PER ORBIT; PEAK IN SUNLIGHT  
2. BASED ON ARRAY ENERGY CAPABILITY; SOLAR-ORIENTED,  $\beta = 0$   
3. 60-% ENERGY STORAGE EFFICIENCY  
4. 30-kWe CONTINUOUS AVERAGE LOAD

Figure 2-4. Typical Solar Array System Peak Load Capability

## 2.2 SOLAR CONCENTRATOR SYSTEMS

The baseline solar concentrator systems consist of two independent 60-kWe modules each having redundant converters. Operation of a single module can supply the initial power required without the need for an additional energy storage system. During MPTS testing however, both 60-kWe modules are operated, and an energy storage system is supplied with the test antenna to store peaking power. This system can supply 450 kWe for 10 to 12 minutes every orbit (or 18 to 20 minutes every other orbit).

If 15 minutes of testing proves to be required for each orbit, the modules could be sized at 70 kWe instead of 60. This would entail no significant additional expense since detail design of these systems has not yet been undertaken. On the other hand, if testing less frequently than every orbit (e.g., every three or four orbits) proves to be acceptable (as seems probable to allow time for assimilation and use of the test data to modify the test program), use of only a single 60- or 70-kWe module during this period should be considered. This would eliminate the need for deploying the growth capability system early in the SCB program. Figure 2-5 gives typical duration peak power profiles for these systems.

CR 75

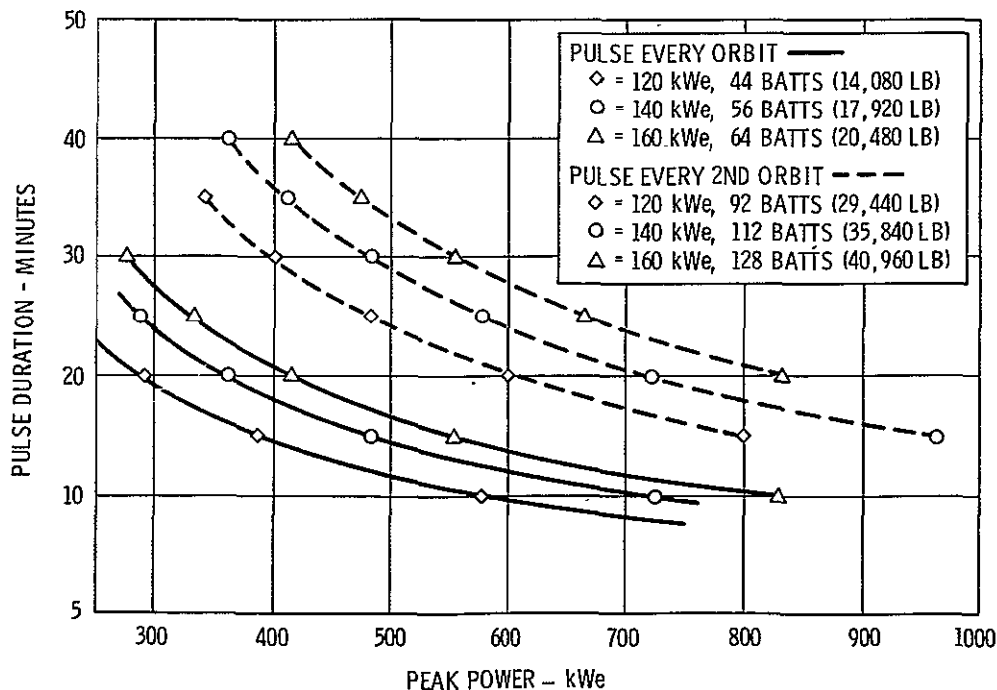


Figure 2-5. Constant Power Source, Balanced Battery Peak Capability — 10-kWe Baseline Load

When MPTS testing is completed, operation using only one 60- or 70-kWe module would be continued until growth power is required by the SCB.

### 2.3 NUCLEAR SYSTEMS

The baseline nuclear systems consist of two independent modules each having three redundant 60- to 120-kWe converters (i.e., the converters may be operated initially at 60 kWe and then 'switched'\* to 120 kWe). This converter growth capability eliminates the need to operate two reactors simultaneously which would result in increased radiation and a requirement for heavier shielding.

One module would be operated at 60 kWe prior to MPTS testing and switched to 120 kWe during MPTS testing. An energy storage system would be provided as part of the test antenna article to store peaking power. Peaking performance would be the same as that for the solar concentrator systems shown in Figure 2-5. The comments (and curves) on performance vs duration and sizing are the same as for the solar concentrator case.

Upon completion of MPTS testing, the excess power produced by the 120-kWe operation would be dissipated in a parasitic load until growth power is required by the SCB.

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\*The brayton engine is "switched" by increasing the helium-xenon charge to match the engines generating capability with the source thermal power increase.

### Section 3

## ALTERNATIVE POWER SYSTEM CANDIDATES

This section of the report covers the initial list of the candidate power systems, requests for additional information, assessments of power system responsiveness to SCB requirements, and selection of two power systems for SCB integration analysis.

### 3.1 CANDIDATE POWER SYSTEMS

The alternative power system candidates for comparison with the baseline solar photovoltaic system are:

- Isotope dynamic (brayton and organic rankine cycles)
- Solar brayton
- Solar thermionic
- Heat pipe reactor brayton
- Heat pipe reactor thermoelectric
- Heat pipe reactor thermionic

Copies of the solar and reactor power system descriptions provided by NASA's Jet Propulsion Laboratory and Lewis Research Center (LeRC) and ERDA's Los Alamos Scientific Laboratory (LASL) are contained in appendixes to this report. The data used for isotope dynamic systems (References 3-1 and 3-2) are not included since the power capability of these systems makes them candidates only for emergency SCB power requirements.

### 3.2 POWER SYSTEM DATA REQUESTS

Following review of the submitted system designs, MDA/C requested additional data for each candidate. First we asked for information on system design redundancy required to provide reasonable probability of attaining the 10-year-life goal for each power system. Then we asked for details of the development program needed by each system to attain SCB launch readiness in 1985. Our approach to meeting the 10-year-life requirement and the rationale for requesting a development schedule are summarized in the following paragraphs.

Each submitted design was optimistic about attaining 10-year system life and none anticipated maintenance requirements. MDAC's integration of reactor power systems with a 33-foot Space Station in the Phase B study addressed long-life requirements for both static and dynamic conversion systems (Reference 3-3). During the referenced study, we analyzed in detail the redundancy required and identified the components/elements which would probably fail during a 10-year mission, based on the best available failure rate data for all system components. It was concluded that minimum maintenance would be necessary, but that maintenance was the only reasonable way of attaining the 10-year lifetime. The individual agencies were requested to redefine their redundancy needs based on similarity to the previous analysis provided them. Revision of the reactor power system configurations was also requested to permit astronaut access for maintenance activities. Acceptance of the maintainability concept was obtained from ERDA LASL and NASA LeRC.

Information was requested on the power system development requirements to gain insight into current development status on major system elements, and to determine remaining development needs and the completeness of the development planning. With this information, MDAC would be in a position to assess the relative development costs and risks. Unfortunately, the data received were neither quantitative nor sufficiently specific to permit comparisons of systems. Since none of the systems has development status, the lack of specific detail is not surprising. Consequently, cost and risk comparisons are qualitative.

### 3.3 POWER SYSTEM RESPONSIVENESS

Space construction base power requirements are variable and pose significant demands on any power system. The key ones are:

- Initial power level — 50 kWe average
- Growth power level — 100 kWe average
- Peak power demand — 20% for 1 hour every 8 hours
- MPTS test requirements — 453 kWe for ~15 minutes each orbit
- Provide 50% power for credible system failures.

The normal growth power requirement can be met either by adding a second power module or by overdesigning the power output of the individual power

module. From a practical point, only the reactor power modules can effectively increase their thermal power output without difficulty. Thus, the solar systems are designed to meet the growth power level by operating a second power module.

Nominal peak power requirements can be met by providing power storage capability; both baseline solar array and thermionic power systems use this approach. However, the solar brayton and reactor brayton systems provide power during both the eclipse and daylight portions of the orbit. These systems therefore satisfy the peak power level by providing continuous output of either 60 or 120 kWe with unneeded power dumped to parasitic load banks.

The most significant SCB peak power requirement is that for MPTS testing. The need for a 450-kWe pulse for 15 minutes each orbit imposes a significant energy storage system requirement and a curtailment of other SCB activities during the test time period. A parametric analysis was made (see Section 2) to assess MPTS test impact on power system sizing. Briefly, all the candidate systems meet the pulse requirement but the test hardware must provide the needed storage subsystem capability. Equivalent pulse power performance was obtained using the following power module sizes plus MPTS battery storage as required.

|                   | <u>Maximum<br/>Power (kWe)</u>         | <u>Average<br/>Power (kWe)</u> |
|-------------------|----------------------------------------|--------------------------------|
| Solar array*      | 250 kWe maximum array power (sunlight) | 100                            |
| Solar thermionic* | 2 modules at 115 kWe (sunlight)        | 100                            |
| Solar brayton     | 2 modules at 60 kWe (continuous)       | 120                            |
| Reactor systems   | 1 module at 120 kWe (continuous)       | 120                            |

\*These systems require a 100-kWe storage/output element

### 3.4 PRELIMINARY POWER SYSTEM ASSESSMENT AND SELECTION

The candidate power systems were assessed based on the system designs provided by Government agencies, and were compared within the solar and reactor heat source categories, to select one candidate from each category for integration with the SCB. The comparisons were based on preliminary integration with the SCB configurations and power system responsiveness to updated SCB EPS requirements.



### 3.4.1 Solar Systems

The comparison of the solar brayton and solar thermionic power systems is shown in Table 3-1. In the areas of concentrator performance and gimbal requirements, the systems are approximately equivalent even though the thermionic concentrator's requirement for allowable mirror surface and pointing accuracy is more restrictive. The solar brayton concentrator is made up of 24 petal segments to be assembled in orbit. In contrast, the solar thermionic concentrator is built up in three layers starting with a deployable antenna base. After antenna deployment, rigid petal segments are attached to the antenna ribs and smaller mirror segments are then attached to the petal base.

In comparing the brayton engine with the thermionic diodes, a marked difference is noted in technology confidence. A 2- to 15-kWe brayton engine has been in operation (unattended) at the Lewis Research Center for more than 30,000 hours, performing at system efficiencies at or above requirements. Scaling to the SCB power requirements will be necessary; however, the engine and heat exchanger technology appears well in hand. In contrast, thermionic diode development has not progressed to a comparable level of accomplishment. Reference 3-4 provides NASA-endorsed performance projections for the SCB period of interest. These projections are shown in Figure 3-1 and indicate a 30- to 40-percent confidence that 20-percent converter efficiency will be available in 1980-1981. The 90-percent confidence curve shows only a 5-percent converter efficiency at the end of 1982.

Comparison of system weights shows a strong difference in favor of the brayton cycle concept, primarily because batteries for eclipse power are required with the basic thermionic power module. The addition of a phase change heat storage capability could result in major thermionic weight savings and concentrator diameter reductions. However, the expected converter performance (Reference 3-4) negates these possible gains and would create significant size increases. Consequently, the solar thermionic system must be considered a high-risk option, thus the solar brayton system is selected to represent the solar category for SCB integration.

Table 3-1  
COMPARISON OF SOLAR POWER SYSTEMS

| Areas of Comparison                                                            | Solar Brayton                                                                                                                                                                                                                                                             | Solar Thermionic                                                                                                                                                                                                                                        |
|--------------------------------------------------------------------------------|---------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|---------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| Concentrator                                                                   | <ul style="list-style-type: none"> <li>• 1.95 kg/m<sup>2</sup></li> </ul>                                                                                                                                                                                                 | <ul style="list-style-type: none"> <li>• 3.5 kg/m<sup>2</sup></li> </ul>                                                                                                                                                                                |
| <ul style="list-style-type: none"> <li>• Surface and Pointing Error</li> </ul> | <ul style="list-style-type: none"> <li>• Requires high surface pointing and accuracy (~15 min and ~15 min)</li> </ul>                                                                                                                                                     | <ul style="list-style-type: none"> <li>• Requires maximum surface and pointing accuracy, 7.5 min (combined)</li> </ul>                                                                                                                                  |
| <ul style="list-style-type: none"> <li>• Fabrication</li> </ul>                | <ul style="list-style-type: none"> <li>• TBD (Fabrication 1.83-m mirror)</li> </ul>                                                                                                                                                                                       | <ul style="list-style-type: none"> <li>• TBD</li> </ul>                                                                                                                                                                                                 |
| Engines/Converters                                                             | <ul style="list-style-type: none"> <li>• Continuing test of 2- to 15-kWe system</li> <li>• Components must be scaled to SCB power needs</li> <li>• Design of heat exchanger and ducting for 40- to 160-kWe engine</li> <li>• Engine confidence approaches 100%</li> </ul> | <ul style="list-style-type: none"> <li>• Lab and in-reactor test of diodes at alternate temperatures and power densities</li> <li>• Assumes 1980-1981 technology</li> <li>• Reference gives 30 to 40% confidence that performance can be met</li> </ul> |
| Receiver/Thermal Storage                                                       | <ul style="list-style-type: none"> <li>• 11-kWe system designed and fabricated</li> <li>• Tests made on 3-tube segment</li> </ul>                                                                                                                                         | <ul style="list-style-type: none"> <li>• 20-kWe receiver analyzed. No detailed design, fabrication, or test</li> <li>• No thermal storage subsystem</li> </ul>                                                                                          |
| Gimbal Mechanisms                                                              | <ul style="list-style-type: none"> <li>• Equivalent</li> </ul>                                                                                                                                                                                                            | <ul style="list-style-type: none"> <li>• Equivalent</li> </ul>                                                                                                                                                                                          |
| Maximum Diameter                                                               | <ul style="list-style-type: none"> <li>• 21 m</li> </ul>                                                                                                                                                                                                                  | <ul style="list-style-type: none"> <li>• 27 m</li> </ul>                                                                                                                                                                                                |
| Weight                                                                         | <ul style="list-style-type: none"> <li>• 7483 kg</li> </ul>                                                                                                                                                                                                               | <ul style="list-style-type: none"> <li>• 13,424 kg (batteries 10,280 kg)</li> </ul>                                                                                                                                                                     |
| Access for Maintenance                                                         | <ul style="list-style-type: none"> <li>• Equipment accessible</li> </ul>                                                                                                                                                                                                  | <ul style="list-style-type: none"> <li>• Requires crane and cherry picker. Relocate batteries to SCB</li> </ul>                                                                                                                                         |
| Thermal Cycling                                                                | <ul style="list-style-type: none"> <li>• Thermal storage systems allow operation during eclipse</li> </ul>                                                                                                                                                                | <ul style="list-style-type: none"> <li>• Diodes see thermal cycle (on-off) each orbit (58,400 cycles/10 years)</li> </ul>                                                                                                                               |
| Risk (Overall)                                                                 | <ul style="list-style-type: none"> <li>• Low to medium</li> </ul>                                                                                                                                                                                                         | <ul style="list-style-type: none"> <li>• High</li> </ul>                                                                                                                                                                                                |

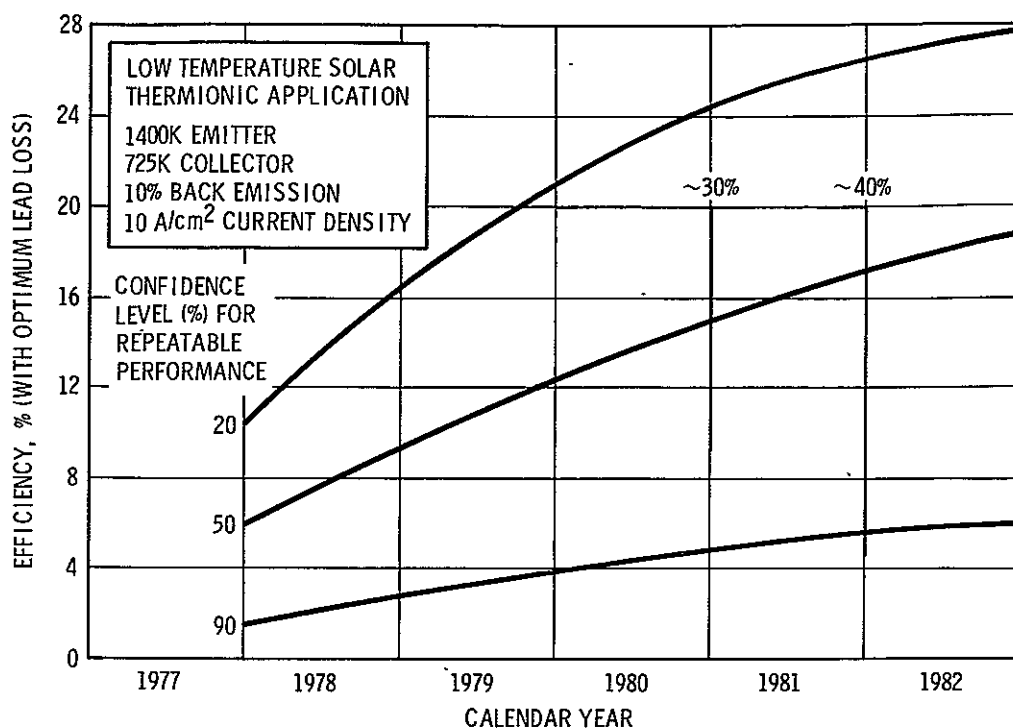


Figure 3-1. Converter Efficiency Projection

### 3.4.2 Reactor Systems

The nuclear power systems all are based on the heat-pipe fast reactor concept developed by LASL for ERDA. Comparisons of the LASL reactor with three power conversion options are given in Table 3-2. Each assumes a conical reactor/shield/structure approximately 2.13 meters (7 feet) long attached to a cylindrical radiator segment within the 16-meter (52.5 feet) of space available in the Orbiter payload bay. The brayton conversion system, with an efficiency ratio of 25 percent, has an electrical power capability of 250 kWe, but would require 353 m<sup>2</sup> (3800 ft<sup>2</sup>) of radiator surface at that power level. For the assumed configuration, the maximum power level will be limited to approximately 135 kWe at a radiator temperature of 475 K.

Three suboptions are shown for the thermoelectric converter system which uses different combinations of thermoelectric materials and radiator temperatures. These systems can use either the silicon-germanium modules with converter efficiencies of 8 to 10 percent (depending on radiator temperature) or a more efficient but less developed selenide converter. The Si-Ge

Table 3-2

## COMPARISON OF REACTOR POWER CONVERSION SYSTEMS

| Areas of Comparison                                     | Brayton            | Thermoelectric     |                    |                       | Thermionic              |
|---------------------------------------------------------|--------------------|--------------------|--------------------|-----------------------|-------------------------|
|                                                         |                    | Si-Ge              | Si-Ge              | Adv Tech <sup>1</sup> |                         |
| Conversion Efficiency                                   | 25%                | 8%                 | 10%                | 15%                   | 20% <sup>3</sup>        |
| Maximum Power Capability with B/L Reactors <sup>2</sup> | 250 kWe            | 80 kWe             | 100 kWe            | 150 kWe               | 400-500 kWe             |
| Radiator Area <sup>4</sup> at Maximum Power Capability  | 353 m <sup>2</sup> | 60 m <sup>2</sup>  | 202 m <sup>2</sup> | 408 m <sup>2</sup>    | 44 to 54 m <sup>2</sup> |
| Maximum Radiator Area Available Without Augmentation    | 190 m <sup>2</sup> | 190 m <sup>2</sup> | 190 m <sup>2</sup> | 190 m <sup>2</sup>    | 190 m <sup>2</sup>      |
| Radiator Temperature                                    | 475 K              | 775 K              | 575 K              | 475 K                 | 975 K                   |

<sup>1</sup> Assumed to be selenide technology.

<sup>2</sup> UC reactor (brayton and thermoelectric) has a 1-MW<sub>t</sub> capability per LASL design.  
 UO<sub>2</sub> reactor (thermionics) has a 3-MW<sub>t</sub> capability per W. M. Phillips, et. al. (Reference 3-5).

<sup>3</sup> Thermionic  $\eta$  of 20% at start of study. Reference 3-4 (dated June 9, 1977) indicates low probability of attaining this efficiency by 1981-1982.

<sup>4</sup> Radiator areas extrapolated from LASL data.

converter is limited to about 80 to 100 kWe of power unless the reactor's thermal capability is increased to 150 percent of current design. The selenide converters have a larger potential power rating (e.g., 150 kWe); however, the efficiency increase is partially obtained by lower radiator temperatures (e.g., 475 K) and it would be necessary to increase the radiator area to meet the SCB power requirement. Conventional techniques of augmenting the radiator require the addition of pumped-fluid loops with additional temperature drop and maintenance/reliability questions due to non-heat-pipe heat transfer.

For SCB application at 120 kWe, the thermoelectric systems are marginal with power capabilities limited. To provide the 120 kWe, it would be necessary to operate two power modules (using Si-Ge converters) at one time. With this approach, two thermoelectric systems could produce a maximum of 160 kWe. While this approach is feasible, it is not competitive with either the brayton or thermionic systems.

The selenide thermoelectric system could meet the 120-kWe power demand, but would require  $\sim 139 \text{ m}^2$  ( $1500 \text{ ft}^2$ ) of radiator area over and beyond that available on the power module's surface. Thus, a deployable radiator configuration would be needed along with a pumped intermediate loop to distribute the thermal load to separate radiator segments.

The thermionic power system considered in the right-hand column of Table 3-2 requires a higher-temperature heat pipe reactor design which has a higher thermal power capability, e.g., 3 MW<sub>t</sub>. Consequently, the thermionic power system has a significantly higher power capability, and with the much higher radiator temperatures the system is not radiator-limited.

The reactor brayton and reactor thermionic systems have greater power growth capability and can meet the power requirements without radiator augmentation. In fact, the thermionic system has rather dramatic power-growth capability. Unfortunately, development funding for the thermionic converters has lagged and there is a higher risk if this system is selected. Reference 3-4 provides NASA-endorsed higher temperature thermionic performance projections for the periods of interest. Figure 3-2 shows these

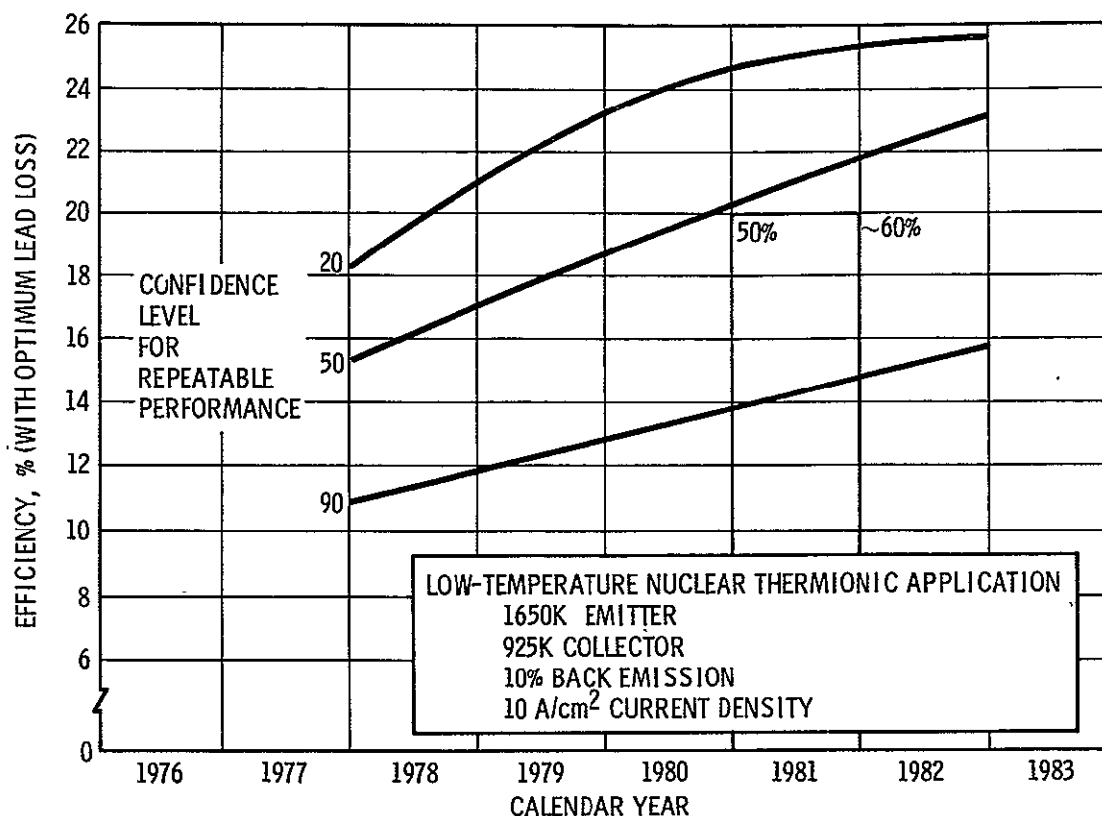


Figure 3-2. Converter Efficiency Projection

projections and indicates a 50 to 60 percent confidence in attaining the 20 percent converter performance at the time of SCB launch. If future mission power requirements exceed the nominal capability obtainable with the reactor brayton system, increased thermionic power development would be advantageous.

Based on the comparisons shown in Table 3-2, the reactor brayton cycle concept is the only system reasonably capable of matching the SCB power requirements by mid-1980 and therefore was carried forward for comparison with the solar brayton concept.

## Section 4

### REACTOR BRAYTON SIZING AND INTEGRATION

The design definition of the reactor brayton power system was provided by Mr. David Buden of the ERDA's Los Alamos Scientific Laboratory (LASL). This section of the report covers the design approach based on his definition, system description, Space Construction Base (SCB) integration, and summary conclusions. Comparisons of the reactor brayton power module with the alternate systems are made in Section 7.

The reactor concept features a high-temperature uranium-carbide fuel in a fast-reactor configuration. High-temperature molybdenum-sodium heat pipes are integrated with the fuel matrix material to provide a design not susceptible to single-point failures in the heat removal element. Elimination of single-point failures in this system represents a major advance in space reactor design reliability. Application of the heat pipe technology has also been extended to the waste heat radiator, eliminating possible radiator failures from single-point penetrations of heat pipe tubes. As designed, the heat source and radiator heat pipes provide built-in redundancy to accommodate multiple heat pipe failures while maintaining the electrical output of the power module. These design innovations are directly attributable to the heat pipe technology developments funded by ERDA at LASL.

The baseline reactor power module design in Figure 4-1 shows the assembled module and lightweight deployment boom. The reactor-shield combination is on the right side of the figure, while the cylindrical segment shown to the left provides the surface area needed for both primary and secondary radiators.

#### 4.1 DESIGN APPROACH

The reactor power module configuration shown in Figure 4-1 will permit astronaut access for maintenance at the black box level; however, the ERDA representative views power system maintenance capability as potentially

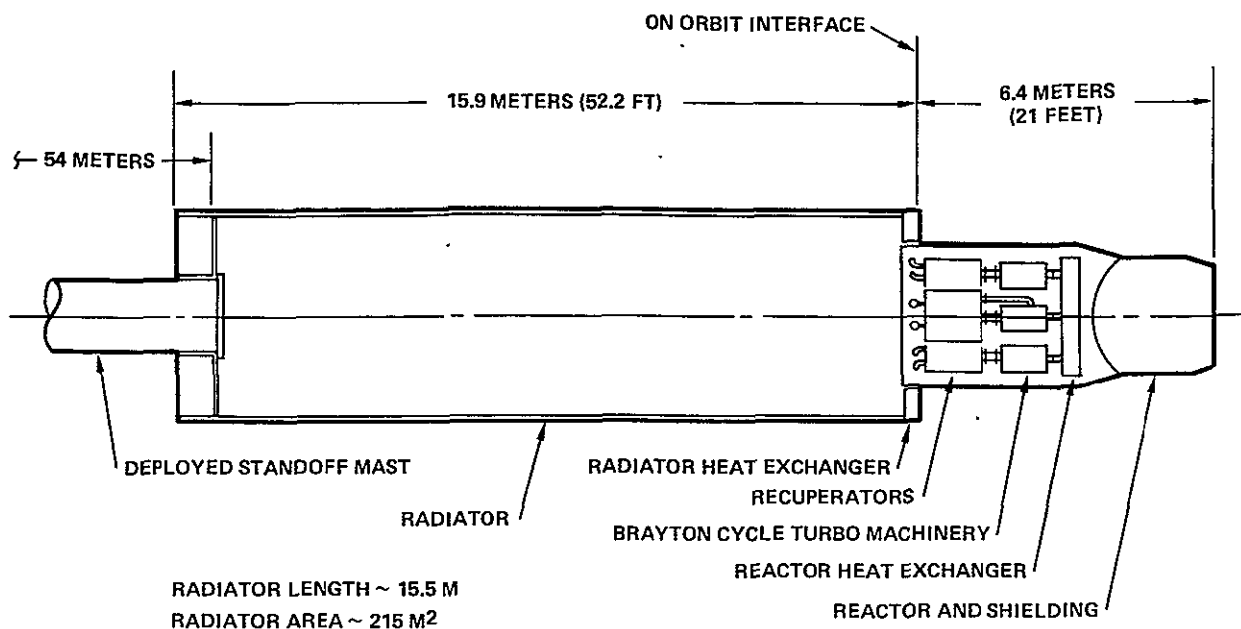


Figure 4-1. Reactor Brayton Power Module - Single-Reactor Launch

degrading to system reliability. Consequently, the baseline reactor brayton power module is not to be maintained. The MDAC reliability/maintainability analysis done during this study shows that power system maintenance is advisable, and that one additional unmaintained reactor power module will be needed to meet the 10-year mission. The unmaintained systems also present a higher program risk. As configured, maintenance capabilities can be provided with minimum design perturbations if ERDA so desires.

An analysis was made to determine the number of power modules to be used and the number of engines to be operated per power module. Table 4-1 summarizes the consequences of operating one or two engines to meet the SCB power requirements. The table also gives the rationale for choosing a dual power capability for each brayton engine. In effect, the variable power capability of the engine is ideally suited to the variable power capability of the reactor. To operate in the dual power mode and to meet the allocated radiation exposure (12.5 rem/quarter), the reactor shield must be sized for the maximum power condition. As a consequence, the radiation exposure rates at half power will be significantly reduced.



Table 4-1  
REACTOR BRAYTON ENGINE SIZING

| Mode of Operation                                                                               | Characteristics                                                                                                                                                                                                                              |
|-------------------------------------------------------------------------------------------------|----------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| Operation of two engines to produce power                                                       | Minimum heat dump problem if one engine fails<br>Increased maintenance<br>Increased redundancy needed<br>Difficult packaging problem, could eliminate double launch<br>Very complex heat source heat exchanger(s)<br>Lower engine efficiency |
| Operation of one engine to produce power                                                        | Maximum engine efficiency<br>Lower maintenance or failures<br>Less complex heat source heat exchanger<br>Easier packaging                                                                                                                    |
| Dual power mode at 60 and 120 kWe                                                               | Operates at 60 kWe with about half the radiation exposure<br>Avoids doubling of power system weight to reach 120 kWe<br>Operation at 60 kWe should make minimum demand on reactor<br>Matches reactor and engine capabilities                 |
| Selection: Operate one engine per module<br>Operate at average + peak power, 60 kWe and 120 kWe |                                                                                                                                                                                                                                              |

Three brayton engines were selected for each power module on the basis of design similarity to that in the 1970 Phase B Space Station/Reactor Power Study; each engine is capable of 60 and 120 kWe operation but would be operated at the 60-kWe level for early missions and 120 kWe for growth missions. The earlier study was based on detailed preliminary power system designs and developed representative component failure rates by work in conjunction with both the reactor and conversion system contractors. This earlier analysis was reassessed, updated, and reaffirmed during the current study within the limitations of the current design concept. On that basis, four modules with three brayton engines per module are indicated to obtain a power reliability of approximately 0.999 for the 10-year mission. Subsequent studies should be made to determine both reliability expectations and a realistic reliability goal for the new reactor power systems. A prerequisite for such a study would be detailed preliminary designs of all major components and auxiliary support equipment.

Operation of multiple engines to provide the growth in power (120 kWe) would dictate additional standby engines to maintain the high system reliability goal.

## 4.2 SYSTEM DESCRIPTION

### 4.2.1 Configuration

Figure 4-2 shows the on-orbit SCB configuration with two reactor power modules installed. As shown, the standby reactor module is retracted and the active system is deployed 100 meters from the manned station modules. Also shown are the Orbiter and a 30-meter radiometer. The latter is typical of mission hardware that will be assembled and tested in low-earth orbit.

Two options were considered for delivery of the reactor power modules. The first, proposed by LASL, was delivery of two complete reactor brayton modules in a single launch. Figure 4-3 shows the dual power module – a single launch with the reactors, booms, engines, and radiators nested together. Analysis of system weights, engine volumes, and Orbiter center-of-gravity constraints indicates this approach is marginal. Consequently, an alternate baseline approach was selected for study purposes. However, the approach of delivering the dual power module in a single launch deserves further consideration after power module design details are developed, particularly for the engines, heat source heat exchanger, radiator, disposal system, and auxiliary equipment. The alternate configuration shown in Figure 4-4 illustrates the design selected for the reactor brayton cycle power module packaged for launch. The deployed module is shown in Figure 4-1.

### 4.2.2 System Performance

The reactor brayton power system schematic is shown in Figure 4-5 for the growth power case (120 kWe). Heat flow from the fission process is via heat pipes at 1,492 K (2,726° F). Individual heat source heat exchangers interface the 91 reactor heat pipes for each of the three brayton cycle engines. Heat will be transferred to the engine working fluid (helium-xenon) when the engine's loop is filled with the gas mixture.

The recuperated cycle is shown in Figure 4-5 with a turbine inlet temperature of 1,273 K (1,832° F). Turbine exit gas passes through the recuperator

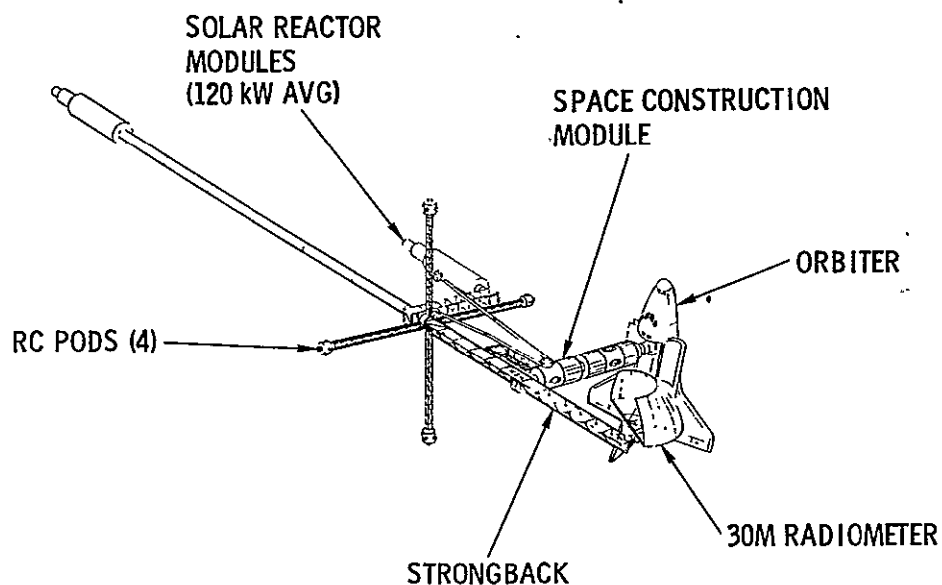


Figure 4-2. Reactor Brayton Power System

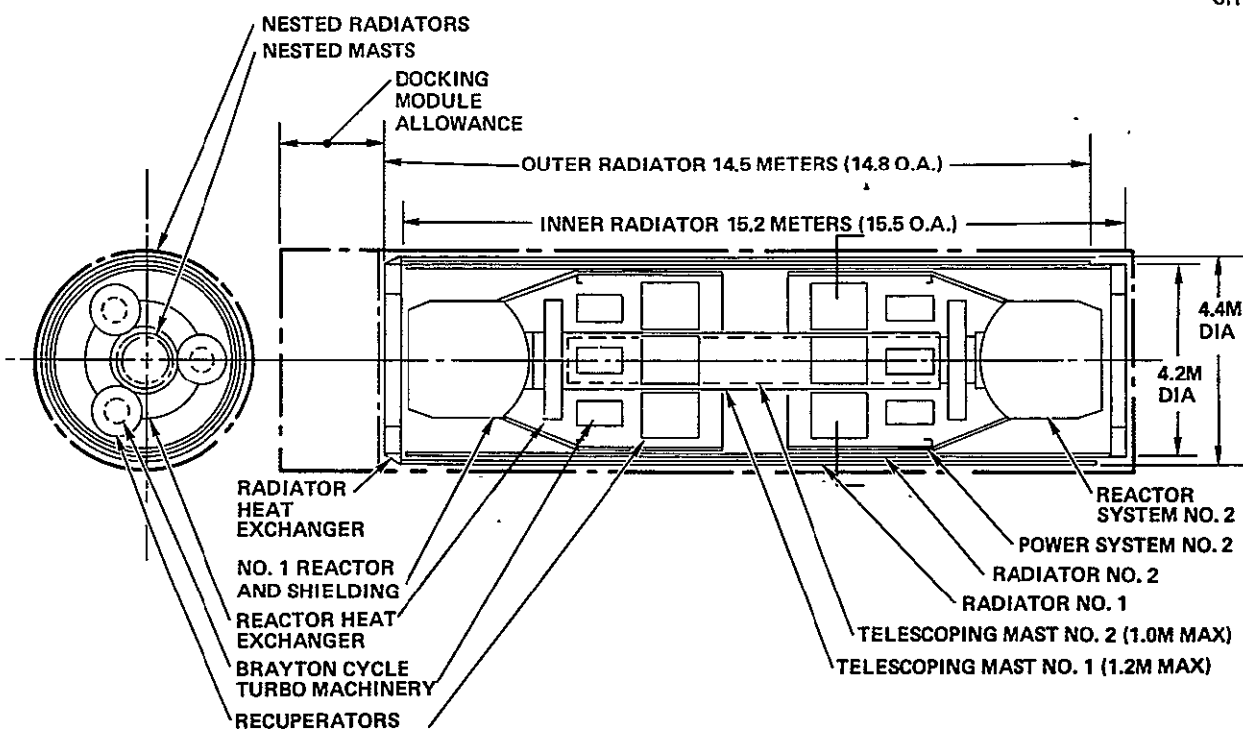
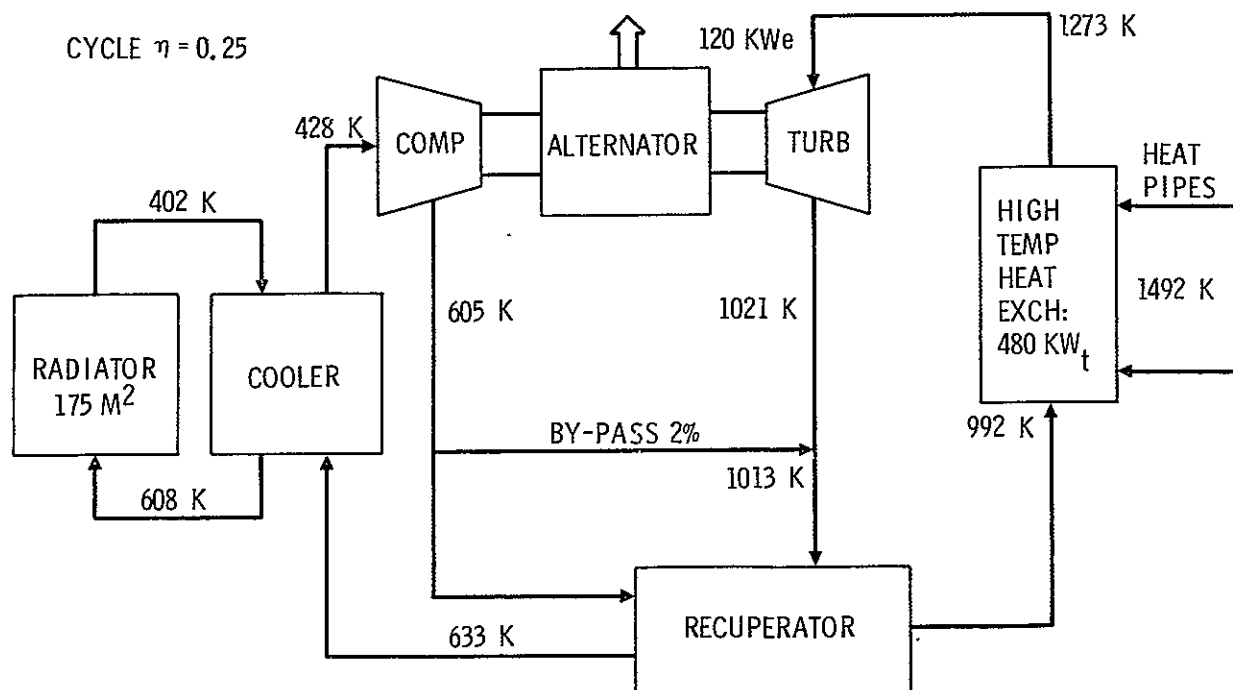


Figure 4-3. Dual Power Module — Single Launch

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**MCDONNELL DOUGLAS**

where waste heat is returned to the cycle as the exhaust flow transfers heat to the gas from the compressor. As the waste heat exits from the recuperator, it is dissipated in the heat sink heat exchanger pumped loop. The latter uses Dowtherm A fluid and distributes the waste heat to the heat pipe radiator elements. After cooling, the helium-xenon enters the compressor and then returns to the heat source heat exchanger by first passing through the recuperator heat exchanger. For the indicated cycle a net thermal efficiency of 25 percent is expected.

Table 4-2 provides summary reactor brayton power system characteristics. It is of interest to note that the LASL shield design, termed a 90-degree shield cone, provides nearly equal shielding over half a hemisphere.

Actual engine speed has not been established; however, indications are that the alternator output will be relatively high-frequency AC power, which will

Table 4-2  
REACTOR BRAYTON CHARACTERISTICS

|                                              |                                             |          |             |
|----------------------------------------------|---------------------------------------------|----------|-------------|
| Reactor                                      |                                             | 325 kg   | (717 lb)    |
| Thermal power used                           | 480 kW <sub>t</sub>                         |          |             |
| Thermal capability                           | 1,000 kW <sub>t</sub>                       |          |             |
| Heat pipes                                   | 91                                          |          |             |
| Shield                                       |                                             | 5,800 kg | (12,790 lb) |
| 90° shield cone                              | ~2 $\pi$                                    |          |             |
| Brayton Engines and Heat Exchangers (3 Sets) | 1,273 K (1,832°F)                           | 2,155 kg | (4,750 lb)  |
| Source Heat Exchanger                        | 1,492 K (2,226°F)                           | 650 kg   | (1,430 lb)  |
| Radiator                                     |                                             | 1,210 kg | (2,670 lb)  |
| Area-Primary                                 | 175 m <sup>2</sup> (1,884 ft <sup>2</sup> ) |          |             |
| Temp (Average)                               | 475 K (395°F)                               |          |             |
| Area - Secondary (Allocated)                 | 40.1 m <sup>2</sup> (432 ft <sup>2</sup> )  |          |             |

be stepped down to meet the SCB requirement for 115/230 VAC 3Ø 400 Hz power.

Table 4-3 presents estimated weights of the reactor brayton power system. Both MDAC-derived weights for specific system or integration hardware and hardware weights that differ from LASL estimates are provided. The areas of difference are detailed in the following section. Weight estimates for launch of one or two reactors per Orbiter flight are given in Tables 4-4 and 4-5.

The single-reactor concept represents the baseline case assumed in this study. As noted in Table 4-5, the two-reactor case is estimated to exceed the Shuttle launch capabilities; however, the weights shown include a 25-percent contingency. If the power system weights and volumes do not

Table 4-3  
REACTOR BRAYTON WEIGHT ESTIMATES

| Element                                                  | LASL   |          | MDAC    |           |
|----------------------------------------------------------|--------|----------|---------|-----------|
|                                                          | kg     | (lb)     | kg      | (lb)      |
| Core                                                     | 325    | (717)    | (325)   | (717)     |
| Shield                                                   | 5,800  | (12,790) | (5,800) | (12,790)  |
| Brayton engines, recuperator and sink heat exchanger (3) | 1,185  | (2,613)  | 2,155   | (4,750)** |
| Radiator                                                 | 410    | (904)    | 1,210   | (2,670)** |
| Structure                                                | 230    | (507)    | (230)   | (507)     |
| Boom*                                                    | 500    | (1,100)  | 500     | (1,100)   |
| Source heat exchanger*                                   |        |          | 650     | (1,433)   |
| Sink/radiator exchanger* loops/pumps                     |        |          | 440     | (970)     |
| Mounting structure*                                      | 910    | (2,000)  | 910     | (2,000)   |
| Disposal package*                                        | 1,475  | (3,250)  | 1,475   | (3,250)   |
| Subtotal                                                 | 10,835 | (23,890) | 13,695  | (30,197)  |
| Contingency                                              | 2,705  | (5,965)  | 3,424   | (7,550)   |
| Total                                                    | 13,540 | (29,855) | 17,120  | (37,745)  |

\*Items for MDAC determination  
\*\*Items of difference

Table 4-4  
INITIAL LAUNCH REQUIREMENTS  
SINGLE-REACTOR BRAYTON

First Launch

|                                                                                            |                       |
|--------------------------------------------------------------------------------------------|-----------------------|
| Power module, boom,<br>attachment structure,<br>disposal package<br>(incl 25% contingency) | 17,120 kg (37,747 lb) |
| Docking module                                                                             | 1,770 kg (3,900 lb)   |
| Total Launch Weight                                                                        | 18,890 kg (41,650 lb) |

Second Launch

|                                                                   |                       |
|-------------------------------------------------------------------|-----------------------|
| Power module, boom,<br>disposal package<br>(incl 25% contingency) | 15,980 kg (35,240 lb) |
| Docking module                                                    | 1,770 kg (3,900 lb)   |
| Total Launch Weight                                               | 17,750 kg (39,140 lb) |

Number of Launches: Two

Table 4-5  
INITIAL LAUNCH REQUIREMENTS  
(TWO REACTORS/LAUNCH)

|                                  | LASL   |          | MDAC   |          |
|----------------------------------|--------|----------|--------|----------|
|                                  | kg     | (lb)     | kg     | (lb)     |
| Two Reactor Power Systems +Booms | 21,125 | (46,580) | 28,275 | (62,346) |
| Attachment Structure             | 1,140  | (2,514)  | 1,140  | (2,514)  |
| Two Disposal Packages            | 3,688  | (8,132)  | 3,688  | (8,132)  |
| Total                            | 25,953 | (57,226) | 33,100 | (72,990) |

grow to the current estimate, it may be feasible to consider a two-at-a-time launch.

#### 4.2.3 Reactor Brayton Components

##### Uranium Carbide Heat Pipe Reactor

A simplified illustration of the fast reactor concept proposed by LASL is shown in Figure 4-6. The reactor concept uses uranium-carbide fuel canned around high-temperature molybdenum heat pipes, which penetrate the core

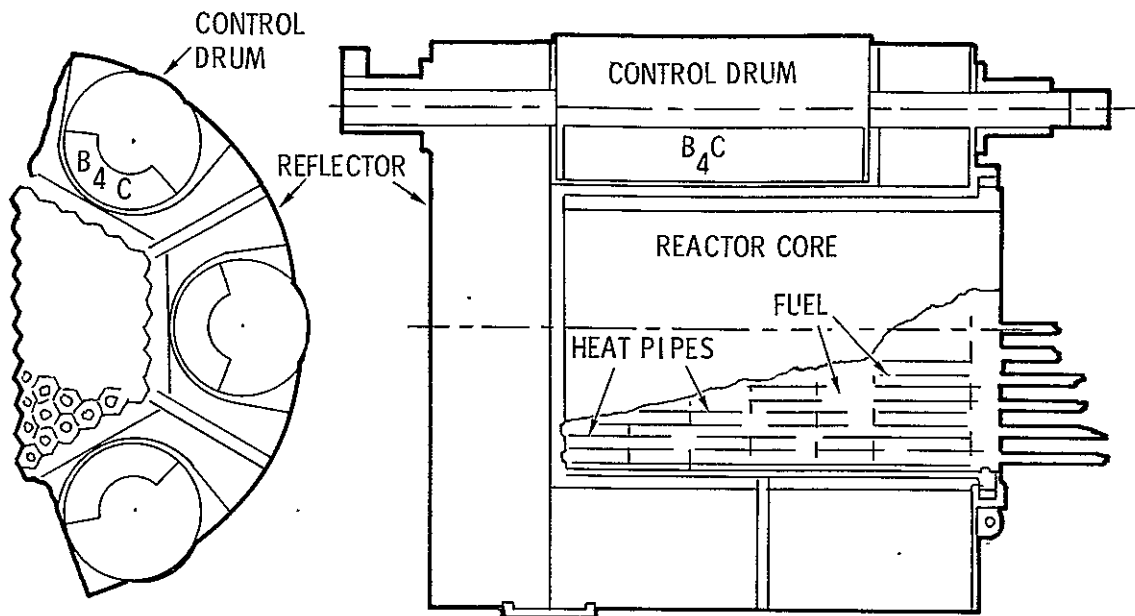


Figure 4-6. Los Alamos Scientific Laboratory Heat-Pipe Reactor

vessel and subsequently interface with the heat source heat exchanger of the brayton-cycle engines. The reactor control elements, reflector, and instrumentation designs were developed from the SNAP and Rover reactor technology bases. Additional descriptive material on the reactor is contained in Appendix D and in Reference 4-1.

The initial configuration provided by LASL, shown in Figure 4-7, gives the general configuration for the reactor, shield, heat source heat exchanger, and the brayton engines.

#### Reactor Shield

A conservative reactor shield configuration has been selected that provides a wide angle of uniform shield thickness, as shown in Figure 4-8. The shield is adapted from earlier development efforts for the SNAP-8 reactor and uses a combination of lithium hydride and tungsten to provide radiation attenuation.

An initial radiation allocation for the reactor power source was established at 12.5 rem/quarter on the basis of MDAC radiation exposure analysis for



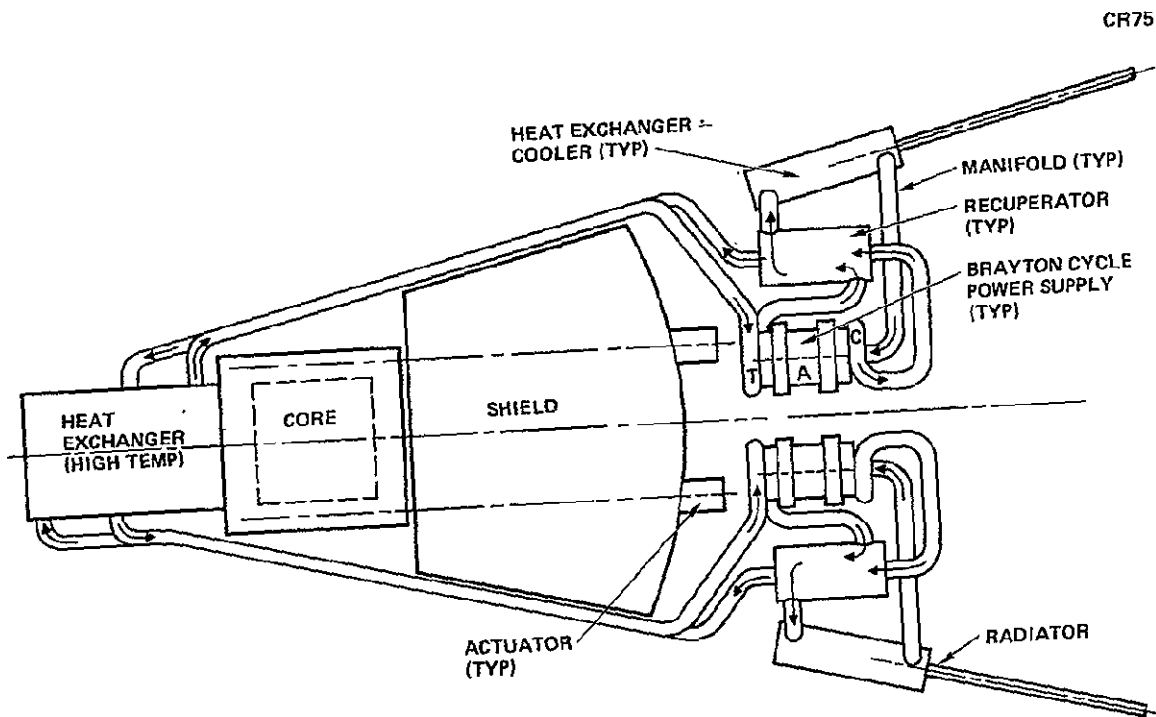


Figure 4-7. Reactor Subassembly With Brayton Converter

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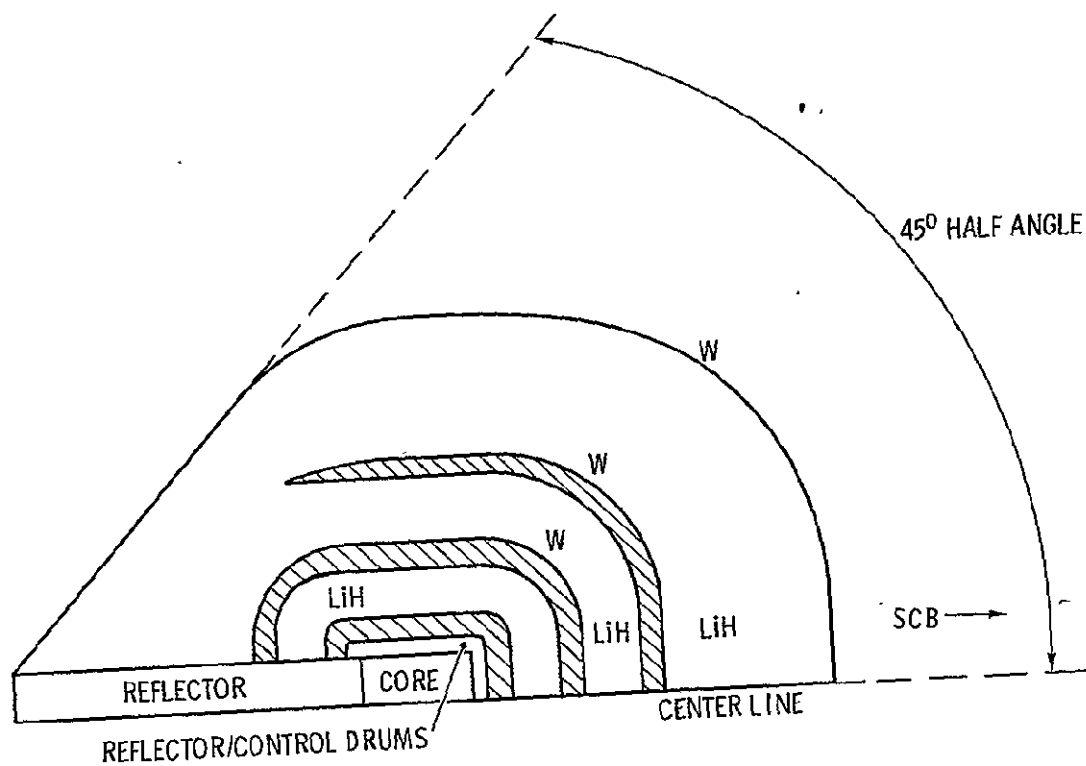


Figure 4-8. Reactor Shield (90 Degrees)

the Phase B Modular Space Station design. Subsequent analysis during this study (see Section 4.3.1) affirmed the dose allocation; moreover, radiation exposure options were prepared as functions of orbit altitude and inclination, dose allowance for extra vehicular activity (EVA), and distance. The new analysis also provides an optimization technique that minimizes the total shield weight for the reactor and the module walls, including the biowell area. (A biowell is a more heavily shielded portion of the module where the crew will reside during a solar cosmic ray event.)

The radiation shield analysis was based on the radiation exposure data provided by LASL for a 450-kW<sub>t</sub> reactor at a distance of 60 meters. Figure 4-9 illustrates radiation shield weight as functions of reactor thermal power and separation distance at the indicated 3-mrem/hr exposure rate. Figure 4-10 translates the LASL data to the baseline system design, giving shield and boom weights for various dose allocations and separation distances. The radiation analysis is presented in Section 4.3.1.

Shield augmentation was examined briefly during the study to determine the requirements inherent in such an approach. The analysis was done to see what gains might result from delivering shield increments in case the reactor

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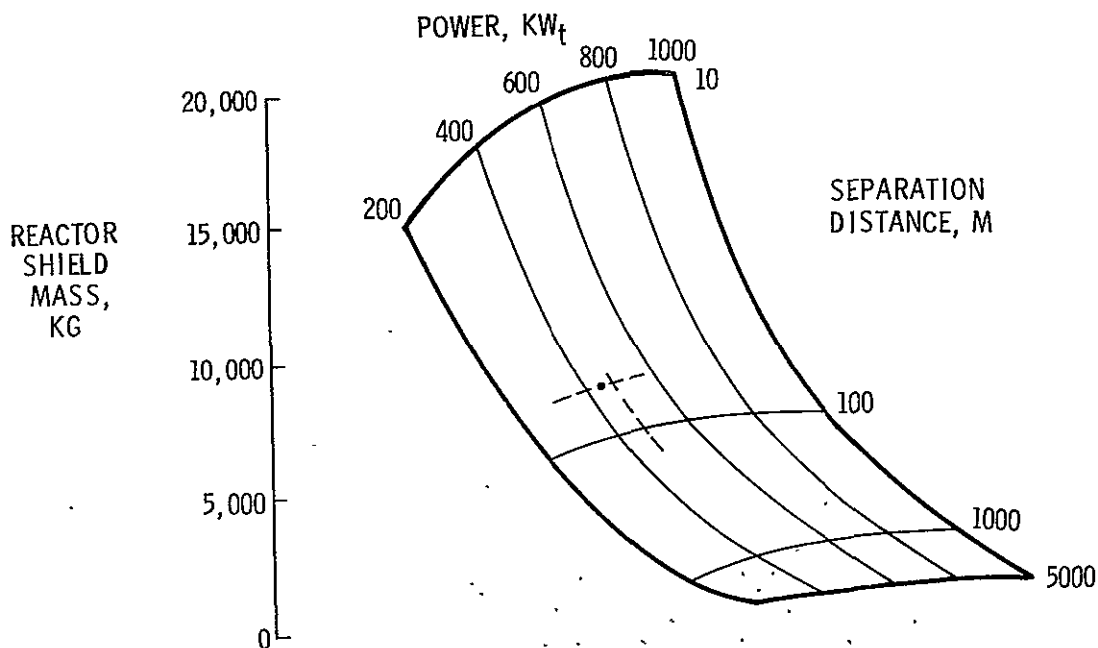


Figure 4-9. Reactor Shield Mass Required (Dose = 3 mrem/hr)

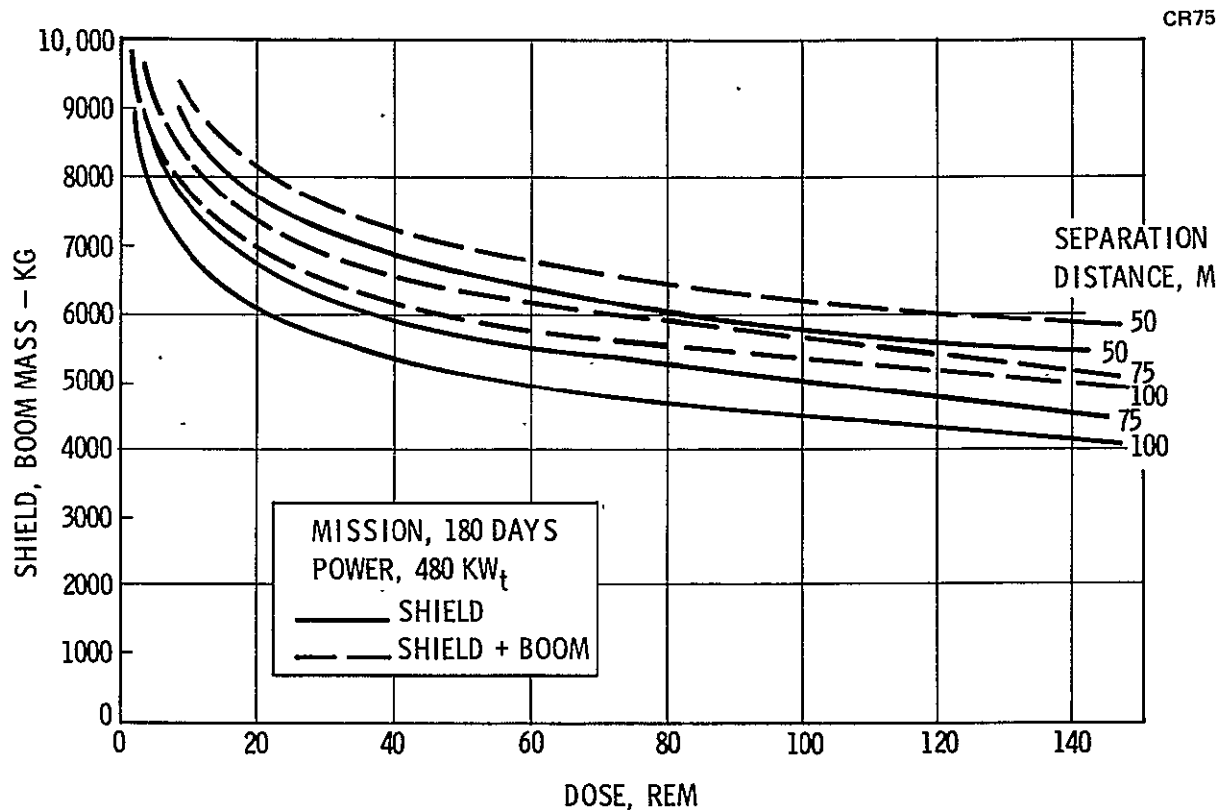


Figure 4-10. Reactor Shield/Boom Mass

power module encountered launch weight problems. The results were as follows:

A. Augmentation Requirements

1. Boom interface with power module and equipment configuration must leave space for shield addition and astronaut access.

Can impact dual launch concept (shortens boom links).

2. Shield additions must be machined/stepped to avoid leakage paths.

Care necessary to maintain lithium hydride between minimum and maximum temperature limits.

3. Reactor shutdown for radiation decay (days) before "close" access is feasible.

TBD impact on astronaut exposure.

B. Alternatives

1. Consider tank/liquid shield - augment during operation.

Leakage risk.

2. If more shielding is necessary,

Can increase shield as necessary - significant unused weight margin.

No apparent need for shield augmentation was found in the radiation analysis, and the augmentation requirements suggest this was a fortuitous result.

#### Heat Source Heat Exchanger

The interface between the heat pipe reactor and the brayton engines is not fully defined; however, one concept developed by LASL for two-engine use is shown in Figure 4-11. At a minimum, the current power module design would require the addition of a third gas-loop heat exchanger. Since the baseline system definition was lacking a weight estimate for this heat exchanger, a conceptual design was developed to establish a preliminary weight estimate, which is given in Table 4-3. (See Section 4.3.4 for heat exchanger sizing.)

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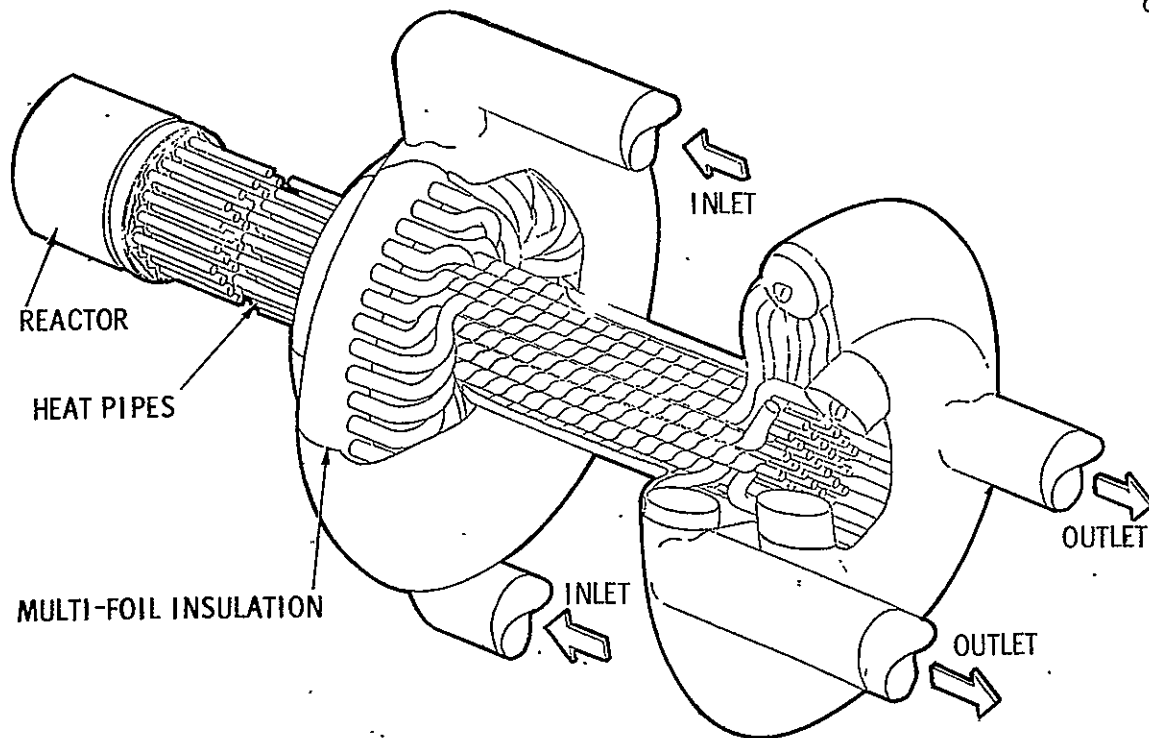


Figure 4-11. Los Alamos Scientific Laboratory Heat Source Heat Exchanger Option

Items of concern in the design of this heat exchanger interface include:

- Fabricability of the entire assembly, including the reactor.
- Reliability of the closely packed multiple gas loops.
- Volume requirements.
- Location of the heat exchanger relative to the shield and SCB.
- Feasibility of heat exchanger redundancy.

- Incorporation of a variable-conductance heat pipe to dissipate reactor decay heat.

The final heat exchanger design may well be quite complex, particularly if redundancy is required in the heat exchanger. A similar reliability question exists for the solar brayton heat source heat exchanger.

Location of the heat exchanger is of concern since penetration of the reactor shield (on the SCB side) could cause radiation to stream toward the station. The total size of the heat exchanger will also be of concern, since added length will definitely create difficulty in packaging two power modules for a single launch.

Figure 4-12 illustrates an LASL design of a variable-conductance heat pipe that can provide for emergency dump of the reactor decay heat in the event the final brayton cycle engine fails. Assuming no heat is removed at the converter interface, heat pipe temperature and pressure will increase, forcing the vapor-gas interface toward the reservoir, and uncovering the emergency heat dump radiator section. Incorporation of this heat dump capability will complicate the heat source heat exchanger design.

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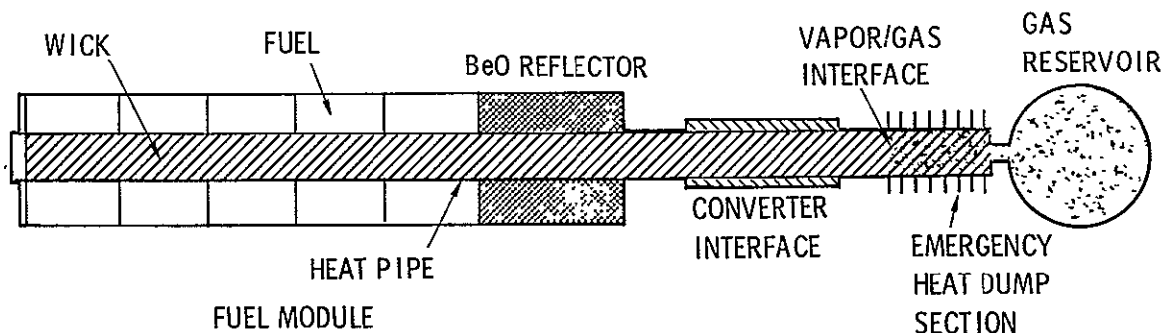


Figure 4-12. Heat Pipe Reactor Emergency Cooldown

### Brayton Cycle Engines

Computer design of the brayton cycle engine has been made for a range of power levels, but specific design details are not available. However, component design and efficiencies for the combined rotating assembly and for the recuperator and heat sink heat exchanger will be comparable to those developed from the LeRC B engine. Differences will exist in the material selected for components that must operate at higher temperatures. At the 1,273 K turbine inlet temperature, the heat source heat exchanger, the turbine and scroll, and the turbine stator must be made of refractory metals such as columbium.

Because of the higher system temperatures, the entire engine assembly will need review to ensure that temperature limits on materials for items such as seals and insulation are not exceeded. Long-term creep limitations on rotating elements also need review.

During the power system analysis, it became evident that brayton engine weight and volume data might be overly optimistic. As a consequence, previous studies were reviewed to determine engine-specific weight trends. Figure 4-13 documents the range of previous Phase B analyses in which an engine contractor provided optimized engine system weights. (Engine system weights include the combined rotating unit, the heat source heat exchanger, recuperator, heat sink heat exchanger, ducting, mounting structure, insulation, and engine control elements.) Also shown in the specific weight curve are the values for the current hardware development (BIPS). It is interesting to note that most of the engine weights fall within the broad band indicated by heavy lines. The dashed line represents the specific weight of engines included in the LASL system design. The black circle at 60 kWe represents the current LeRC weight estimate for the solar brayton engine, which is based on the brayton B engine currently operating at the Cleveland facility. The "expected range" indicates more realistic engine weight estimates, and the upper limit is that assumed by MDAC in Table 4-3.

During this study, support of an engine contractor was sought to clarify the apparent discrepancy; however, a funded design effort will be necessary to arrive at a detailed engine definition.

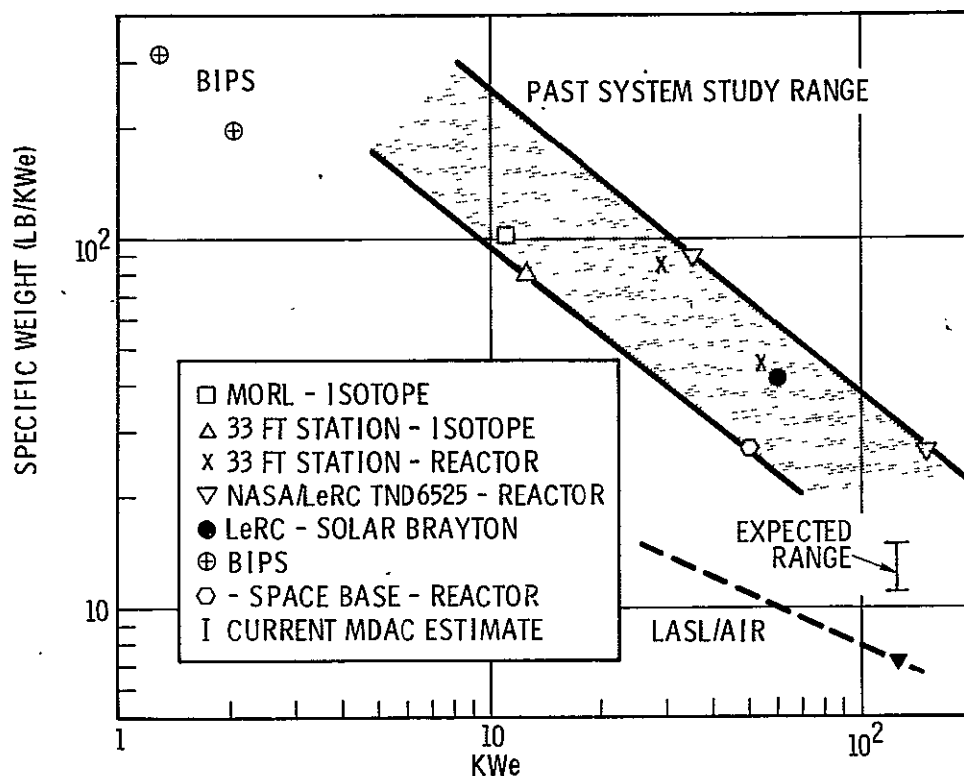


Figure 4-13. Brayton Engine Specific Weights

Similar efforts were made to derive a meaningful specific engine volume curve, but the results were inconclusive. The most recent NASA-funded study was made for NASA LeRC (1970) on a 40- to 160-kWe system (Reference 4-2). The NASA program manager indicated that the engine package probably was considerably oversized to provide a design margin. However, the nominal engine package (at 160 kWe) was approximately 3.0 by 3.0 by 1.4 meters (~10 by 10 by 4.5 feet). Consequently, the concept of utilizing two full-power modules in a single launch is questionable from the viewpoints of volume and weight. Resolution of the discrepancy is beyond the scope of the current study.

The engine interface between the heat sink heat exchanger and the heat pipe radiator also requires further definition. MDAC has made a conceptual design of a redundant pumped-loop interface that uses Dowtherm A to transfer cycle waste heat to the heat pipe radiator. The estimated weight for this

system has been included in Table 4-3. Table 4-6 provides summary data on the heat transport loop and a weight estimate for a heat pipe radiator system.

Table 4-6  
RADIATOR AND HEAT SINK DISTRIBUTION SYSTEM

| Item                                 | Weight        |         | Remarks                        |
|--------------------------------------|---------------|---------|--------------------------------|
|                                      | kg            | (lb)    |                                |
| Distribution Tubes (2)               | 52            | (114)   | 4.37 cm ID stainless           |
| Dowtherm-A Fluid (2)                 | 120           | (264)   |                                |
| Heat Pipe Interface (2)<br>Manifolds | 223           | (492)   |                                |
| Pump Assemblies (4)                  | 45            | (100)   | 447 watts/loop                 |
| Radiator Shell                       | 276           | (609)   | 0.51 mm Al                     |
| Heat Pipes                           | 252           | (555)   | 150 tubes, 1.9 cm ID stainless |
| Heat Pipe Fluid                      | 680           | (1,500) | Mercury, 4.54 kg/pipe          |
| Totals                               | 1,648 (3,634) |         |                                |

Notes: Weights do not include structure, shell only

Distribution and heat pipe tubes are 0.51 mm wall thickness

Radiator area - 200 m<sup>2</sup>

Redundant distribution and pumps

The heat transport loop design is discussed in Section 4.3.4; however, the current concept appears to be quite complex, with fluid heat exchange occurring with each of the 150 heat pipe tubes. The design will merit subsequent review to determine whether or not it can be simplified.

#### Radiator

As noted above, the heat pipe radiator concept has been given a preliminary definition in this study, and the hardware weight estimates appear in Table 4-6. Mercury was selected as the working fluid appropriate for the temperature range of the radiator. It is evident from the weight statement that an alternate fluid would be desirable. The total radiator weight, including the Dowtherm distribution system, is approximately twice the



weight of a normal pumped radiator loop system with redundant loops and pumps. It appears that even with an alternate heat pipe fluid, a conventional radiator system will have a weight advantage.

In addition, the proposed design with redundant heat transport loops and heat transfer capability to each heat pipe may prove to be significantly less reliable due to the multiple piping welds required. Further study is needed to assess the radiator net reliability, feasibility of manufacture, and cost.

#### 4.2.4 Power System Configuration

This section treats the packaging and deployment concepts for the reactor brayton power system modules when integrated with the SCB.

Two configurations are presented: the first places two complete systems in one STS launch, while the second launches each system separately. Weight and volume data developed toward the end of the study have made the dual-module-in-a-single-launch approach questionable with respect to Orbiter capability. However, if the power system can be developed without use of the full contingency weight allocated, this can be an attractive launch mode.

The configuration of the SCB for power system delivery and activation is assumed to occur during the early Shuttle-tended phase rather than in a later phase. Figure 4-14 shows the facility configuration for this phase. It consists of a construction support module, an advanced long-reach crane, a construction platform strongback, an attitude control system, and a 38-kWe solar array power system. This facility is a construction operations complex, and cannot be utilized without the Orbiter for crew and other subsystem support or functions.

The advanced power system concept shown in Figure 4-15 includes one active reactor brayton module and one standby, telescoping standoff masts, and a truss structure for supporting the modules on the SCB. The truss support beam is asymmetrically located on the end of the construction strongback. This asymmetrical geometry simplifies installation of the power modules by crane and tends to reduce resonant coupling between power system modules through the support beam. Analysis, however, will probably show a

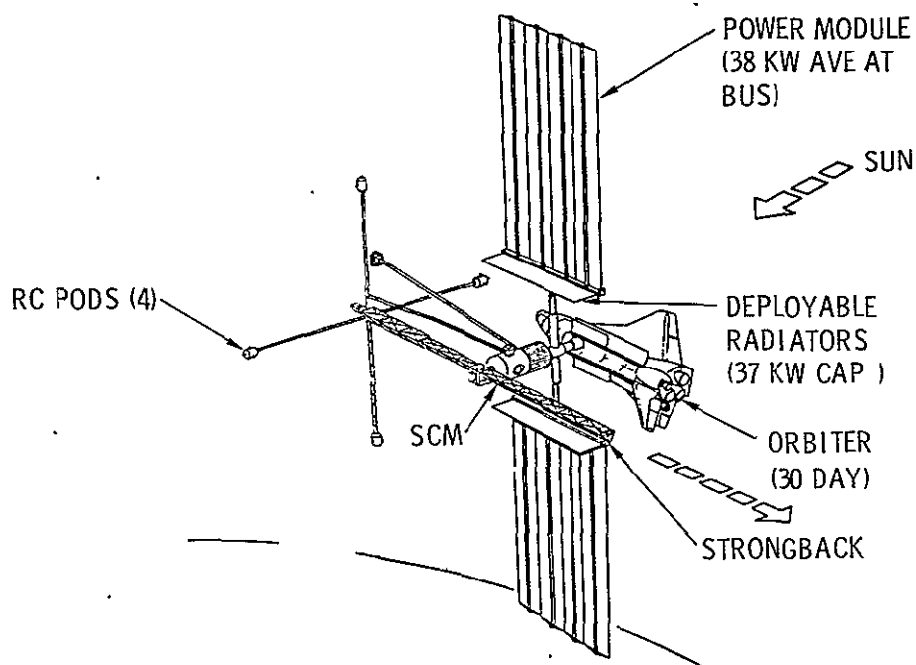


Figure 4-14. Shuttle-Tended Space Construction Base

minimum difference between asymmetrical or symmetrical mounting geometries, and the final choice should be based on facility center-of-gravity preferences for attitude control and orientation ease. Since only one reactor system will be in operation at a time, the active one will be deployed while the inactive one will be fully retracted. The extendable mast will be sized so that the active reactor is nominally located 100 meters away from the facility centerline.

The SCB has an advanced, highly dexterous, two-arm crane with a 35-meter reach. This crane will have sufficient capability to extract the stowed power modules from the docked Orbiter and move them to the construction strongback for intermediate erection and assembly or move them directly to their installed positions on the support beam.

Separately Launched Power Modules

The configuration of the reactor brayton power module for separately launched modules is shown in Figure 4-1. It consists of (1) a power assembly containing the reactor, brayton cycle turbo machinery and various elements of the thermal loops and supportive structure, (2) a radiator, and (3) a telescoping standoff mast. This operational configuration can be compared with the module's as-launched arrangement, shown in Figure 4-4. Orbiter delivery to the SCB requires that an Orbiter docking module be part of the payload. This leaves a usable payload envelope length which, with end clearances, will allow hardware 15.9 meters long. A payload diameter of 4.4 meters is the acceptable limit.

The power system components are packaged so that the radiator uses the full payload bay geometric constraints and is 4.4 meters in diameter and 15.9 meters long. The active length of the radiator is reduced slightly to 15.6 meters to allow for a heat exchanger, which interfaces the radiator with the power equipment. The radiator would be an aluminum shell with longitudinally oriented heat pipes for a passive heat rejection surface. It would have major hoop stiffening structure at each end for heat exchanger and the mast support, as well as minor hoop stiffening along the radiator's length. The heat exchanger will incorporate structural and fluid interfaces with the power equipment segment, which will be made after the reactor and power equipment are extended through the radiator shell during assembly. A simple, lightweight truss-and-rail structure within the radiator is used to support the power equipment module during boost and to guide it during deployment. At the other end of the radiator, a light cruciform beam provides the structural interface between the standoff mast and the radiator.

A cylindrical shell structure supports the reactor shield assembly, the source heat exchanger, three brayton cycle turbo generators with their required heat exchangers, and all of the fluid lines connecting this equipment. To deploy this power segment on-orbit requires a simple extension along the centerline for a distance of approximately 7 meters. Subsequently, the structural and fluid interfaces with the radiator are established.

The standoff mast is a telescoping assembly consisting of six nested segments, each approximately 9 meters long, with a deployed length of 54 meters. With the other structural elements involved, the mast provides a 100-meter standoff. The cylindrical mast sections will be made of graphite-epoxy material to achieve a high stiffness, which maximizes the natural frequency of the long boom-supported mass. The natural frequency is expected to be in the range of 0.01 to 0.02 Hz, which is acceptable for the anticipated control frequency of the SCB. This is somewhat similar to the natural frequency of other candidate electrical power systems such as the 300- to 500-kWe solar cell array concepts. A system of cables would be used to either extend or retract the mast. The mast is nominally 1.5 meters in diameter and when extended will provide a safe and easy EVA tunnel for access to the power system module.

The various crane manipulations and steps required to implement the complete power system configuration for this approach are shown in Figure 4-16. This approach requires two Orbiter launches to get both the active and the standby power system modules on the facility. The Orbiter is docked to the construction support module and the power system module is rotated out of the payload bay by the Orbiter's payload installation and deployment aid (PIDA). The SCB's crane attaches to the power system module and moves it from the PIDA to a temporary holding fixture on the construction strongback. At this location the truss support beam is withdrawn from its stowage position in the standoff mast. The truss box beam has folding struts to allow it to be collapsed into a compact cylindrical envelope. The truss is erected, and interface rings, also stowed in the radiator cavity, are installed.

The support beam is then installed on a mounting ring on the attitude control system centerbody. Next, the power equipment segment is deployed and joined to the radiator and structure. Then, the module is moved by the crane to its mounting location on the support beam. After subsystem interfaces are made and verified, the mast is extended and the reactor startup begins. The second power system launched can be transferred from the Orbiter to the support beam for direct installation and assembly.

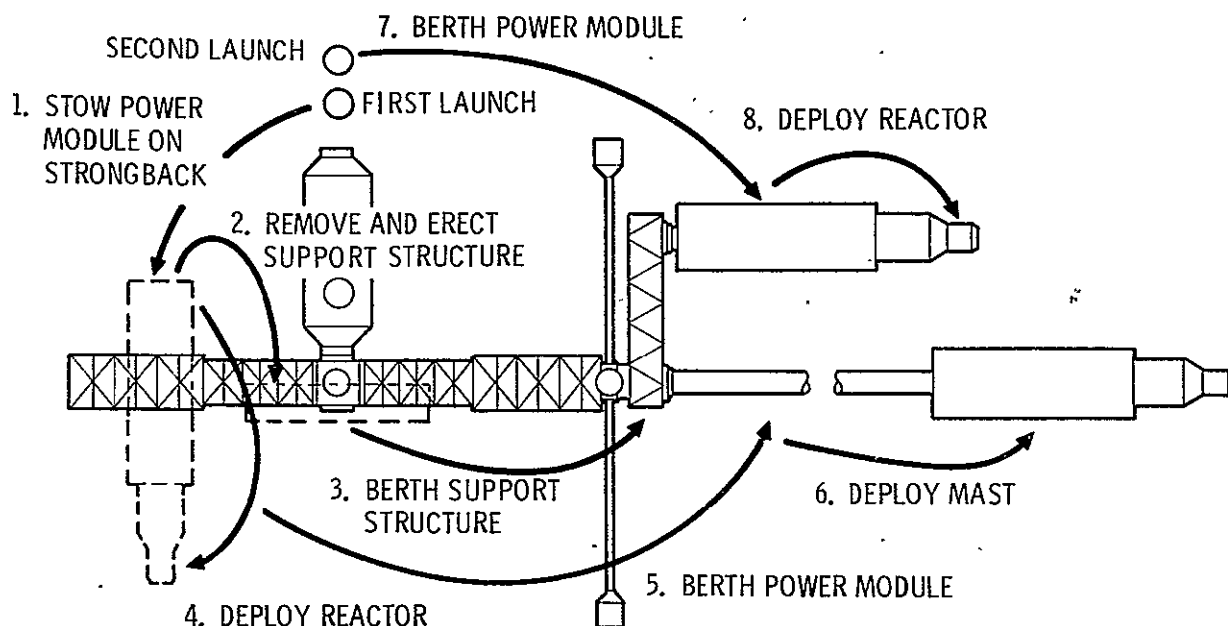


Figure 4-16. Independently Launched Power Modules Build-Up Sequence

#### Single-Launch, Two-Reactor Power System Concept

The configuration for launching two complete reactor brayton cycle modules at one time is shown in Figure 4-3. In this arrangement, the standoff mast mounts directly to each reactor shield assembly. This necessitates alternate geometry for the power generation and thermal equipment to allow mast positioning inside of the engine array. The overall geometry of the power system module in its operational configuration is nearly the same as in the separate launch concept defined in the preceding section.

For the two radiators to nest, they must be fabricated to different diameters. Both radiators could be sized to the maximum area permitted by this arrangement, but the resulting difference in effective area might require slightly different operation power levels. The radiators shown in Figure 4-3 have been sized to a common effective area, which is dictated by the limits of the inner radiator. Aside from these minor variations in geometry, the characteristics and materials would be similar to those defined

in the preceding section. (The unique fabrication requirements are reflected in the power system cost analysis.)

The power equipment segment is virtually the same as that defined for the separately launched concept except that it is larger in diameter to allow the standoff mast to be attached to the reactor shield body. This is the maximum size that will fit in the radiator shell, and will require removal of the reactor power segment from the base of the radiator, and manipulation to its mounting interface with the radiator heat exchanger. In the launch configuration, both power equipment modules are supported within the innermost radiators.

In this configuration, the nested masts must be smaller to fit between the machinery, and are nominally 1 meter in diameter. Because the masts extend to the reactor shield, their extended lengths are approximately 75 meters and consist of nine sections, each approximately 8.5 meters long.

The materials and other characteristics are similar to the mast for the separately launched module.

This concept requires only one Orbiter launch; however, a recent weight analysis indicates that the dual launch may exceed the nominal Orbiter performance capability for the appropriate orbit. Moreover, the symmetry of this launch configuration presents an adverse center of gravity for Orbiter launch constraints, and would require several thousand kilograms of ballast weight to meet Orbiter requirements. Although many missions can use excess Orbiter reaction control system (RCS) propellant for center-of-gravity control, significant ballast would still be required. Figure 4-17 shows the weight of the combined launch, the right curve indicating the weight limit for a 16.2-meter (53-foot) module to be about 24,040 kg (53,000 lb) to accommodate the necessary ballast at Station 1300.

The operational sequences for buildup of a single-launch, two-reactor configuration are given in Section 4.3.2.

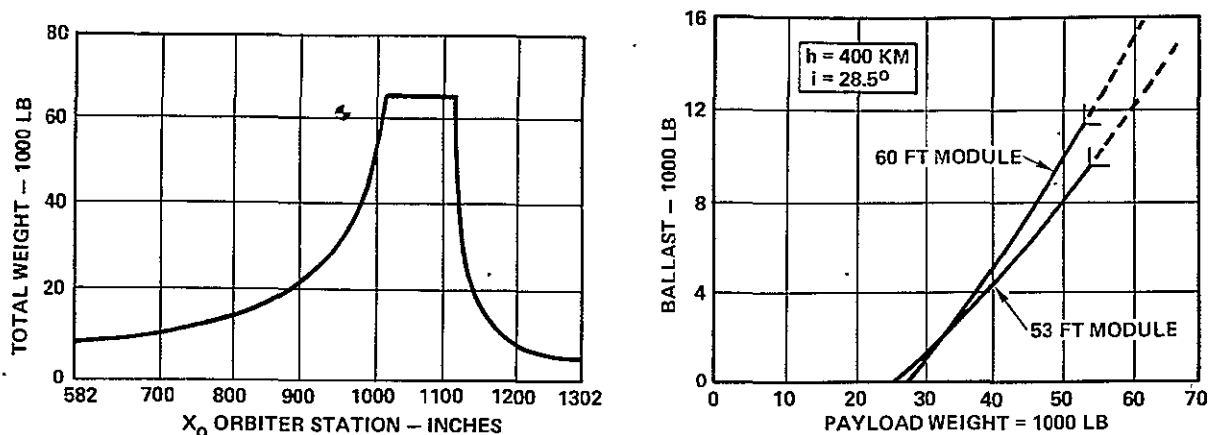


Figure 4-17. Shuttle Center-of-Gravity Constraint

#### 4.3 INTEGRATION

The reactor brayton power system has been assessed in terms of the combined space and nuclear source radiation environment, SCB operations, guidance and control, and thermal environment. In addition, a special analysis assessed power module reliability so that confidence could be gained that this system has prospects of a lifetime equivalent to that of the others. The reliability/maintainability analysis is also included in this section.

The main integration effort was directed at the Shuttle-tended SCB configuration given in Figure 4-14, which shows a 38-kWe power module providing electrical services. This view illustrates the hardware elements on-orbit and available for buildup of either the solar array power platform or the reactor brayton system. After in-orbit assembly of the SCB power system, the power module will be released to function as needed in other Shuttle-supported missions.

During buildup of SCB power systems, the hardware elements will provide electrical power, the services of the construction crane, and the construction capability of the strongback fixture. For this mission definition, a single power module can be assembled, installed, and operated with minimum risk until a second power module can be launched, assembled, and installed.

An alternate mission mode has also been examined to determine what the incremental costs and requirements would be if the power module were launched as an initial free-flyer with assembly done by the Orbiter crew using the remote manipulator system (RMS). This mission option assumes the power system can be operated before launch of any SCB elements and eliminates the need for the 38-kWe power module shown in Figure 4-14.

These two mission modes were assessed by the same personnel who have developed the operational timelines for fabrication, assembly, and deployment of the SCB configurations, mission hardware elements, and SCB baseline power platform; consequently, the comparative analyses are consistent in approach, with each mode considered on the basis of success-oriented operations. The results are documented in the following section, with unique requirements and impacts of the free-flyer power module identified.

#### 4.3.1 Radiation Analysis

The radiation analysis determined the effects of integrating a reactor power system into an SCB in terms of additional shield requirements and reactor dose allocations.

The key assumptions made were:

- A. Natural electrons - AE-5 and AE-7
- B. Natural protons - AP-5, AP-6, and AP-7
- C. Solar cosmic ray event - November 12, 1960 flare
- D. 480 kW<sub>t</sub> reactor data - Derived from Los Alamos data
- E. Reactor dose =  $f(\text{distance squared})^{-1}$
- F. 90-degree reactor shield mass =  $f(\text{thickness cubed})$
- G. Module and body shielding ignored for reactor dose.



First the SCB radiation environment, dose, and shield requirements without a reactor were analyzed, and then with the reactor added to determine any required changes. The analysis considered orbit altitude, inclination, duration, EVA exposure, critical body organs, module and EVA suit designs, and reactor distance and shield thickness. SCB shield weight minimization programs were altered to include the reactor effects. Detailed analyses of the station radiation problem, without the reactor, are reported in Reference 4-3.

The skin, eyes, and bone marrow doses received inside an 0.1-inch-thick wall SCB module from trapped radiation are shown in Figure 4-18 along with the respective 90- to 180-day mission allowables. These numbers have been factored to daily dose numbers for convenience. As shown, the low-orbit inclination dose (at 28.5 degrees) is well below the allowable for the 400- to 500-km altitude range of interest. The unused allowable dose could then be

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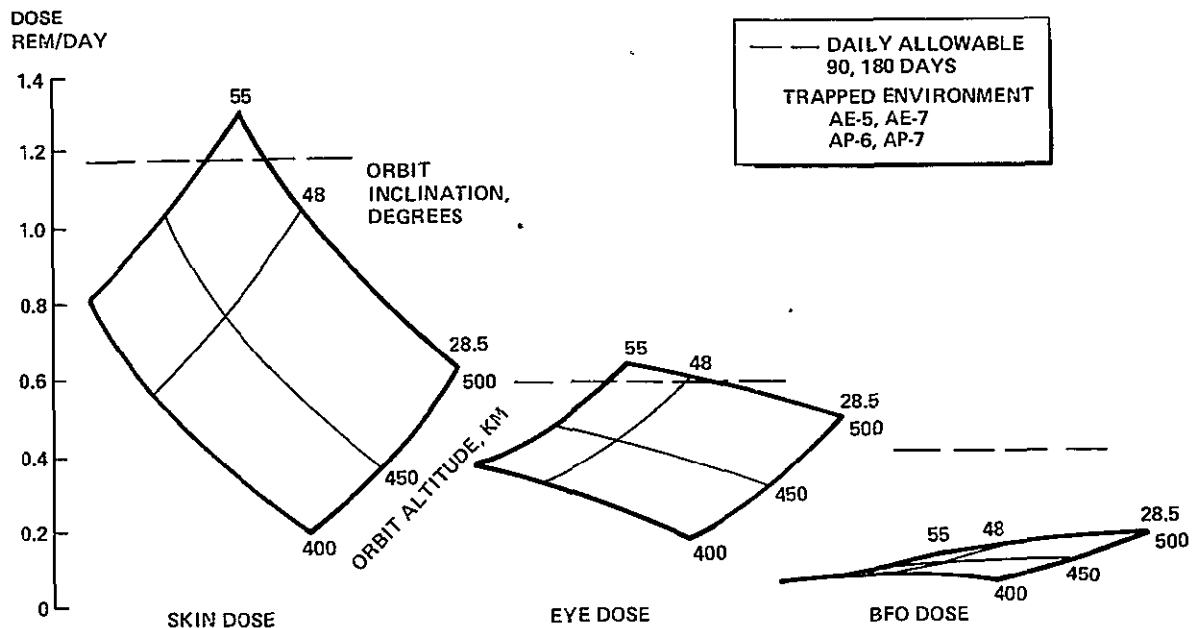


Figure 4-18. Module Radiation Dose - 0.1-Inch Wall Thickness

allocated to other dose sources such as EVA activities and reactor power systems. The solar cosmic ray dose is negligible at 28.5 degrees, due to earth's magnetic field shielding. At this inclination, the addition of a reactor radiation dose would be limited by the blood-forming organs (BFO) dose, because the unused allowable for BFO is less than the allowable for eyes or skin and the reactor radiation is more penetrating.

At 55 degrees the allowable dose for skin and eyes is exceeded at higher altitudes by the background radiation only, as indicated in Figure 4-18. The addition of EVA activities, solar cosmic ray events, and other radiation sources such as a reactor would thus impose a requirement for additional shielding at 55 degrees. The amount depends on how the allowable dose is allocated to each radiation source. This was done in a manner to minimize the total shield requirements.

The relationship of the reactor shield mass to dose, which was derived from the reactor definition input data, is shown in Figure 4-10. It was derived by correlating data on dose, shield thickness, shield mass, and power level, assuming that the dose was proportional to the separation distance squared and the shield mass proportional to the thickness cubed. The separation boom masses were added to consider total weight penalties. These dose data were then added to the natural radiation dose data at both 28.5- and 55-degree inclination missions.

The 28.5-degree orbit shield case is summarized in Figure 4-19. The common ordinate is bone marrow dose, with an allowable for 90 days equal to 35 rem. The right side of the figure shows the dose received within an 0.1-inch-thick wall module and that received inside the Shuttle EVA suit ( $0.1\text{-g/cm}^2$  thickness) doing the EVA necessary for the construction program (six 6-hour shifts per week). These are subtracted from the 35 rem allowable to leave the yet unallocated dose as a function of altitude. The left side of Figure 4-19 is the reactor dose received as a function of shield weight and separation distance. The mounting booms were included to give a total weight penalty. These were obtained by analysis of the reactor data base received from Los Alamos during the study. The common ordinate of Figure 4-19 allows the unused allowable dose at varying altitudes to be allocated to a varying reactor shield weight. At 12.5 rem, the reactor

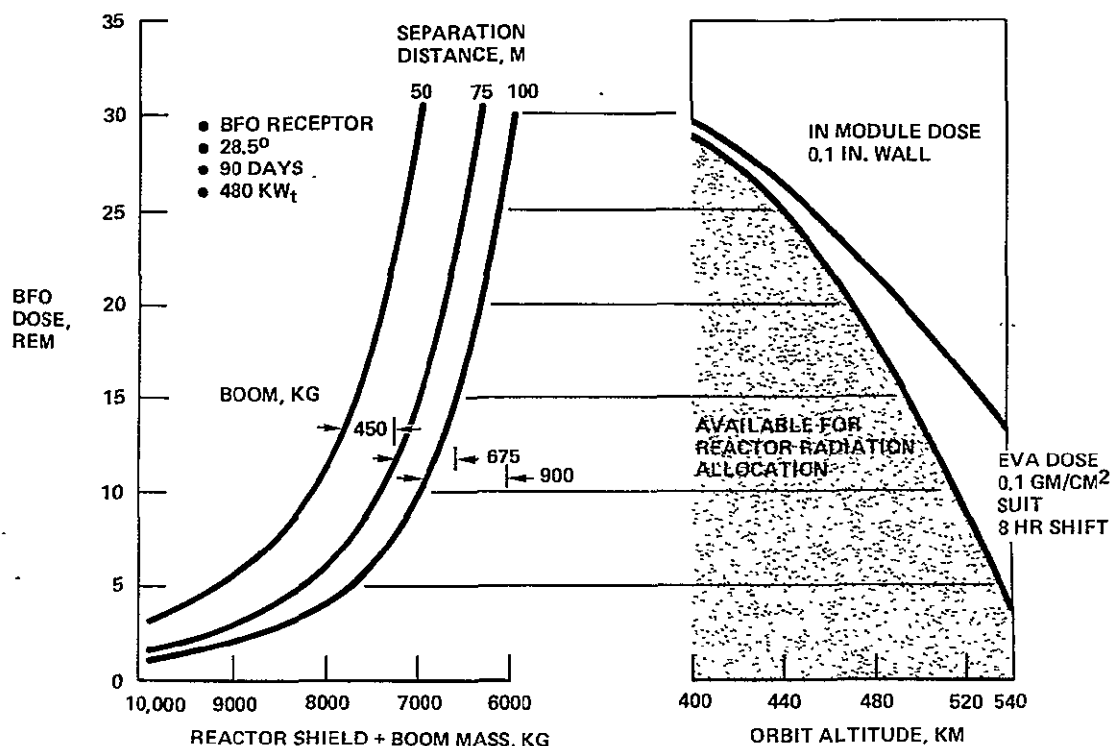


Figure 4-19. Reactor System Dose Parametrics

shield required is 5,800 kg (6,700 less the 900-kg boom). This would allow operation up to 500-km altitude without increasing the module or EVA suit thickness at all. At lower altitudes a higher dose could be allocated to the reactor, and the shield could be reduced. Thus, at 28.5 degrees the reactor can be integrated into the Space Station without penalty.

At 55 degrees, the allowable astronaut dose is received without the reactor. A balance between the required additional module and biowell shielding was calculated to minimize the total shield addition. The addition of the reactor would mean the allowable dose would need to be reallocated to maintain minimum total weight. This was done as shown in Figure 4-20. The total shield mass for the module, including the biowell, and the reactor is shown as a function of reactor and EVA dose allocation. These data are for a 55-degree inclination and a 450-km-altitude orbit of 90 days' duration. For small values of reactor dose allocation, the module and biowell shield required is low, but the reactor shield mass is high. As the reactor dose allocation increases, the situation becomes reversed. A minimum occurs at about 12 rem for a 50-rem EVA allocation. It is about 18 rem for a

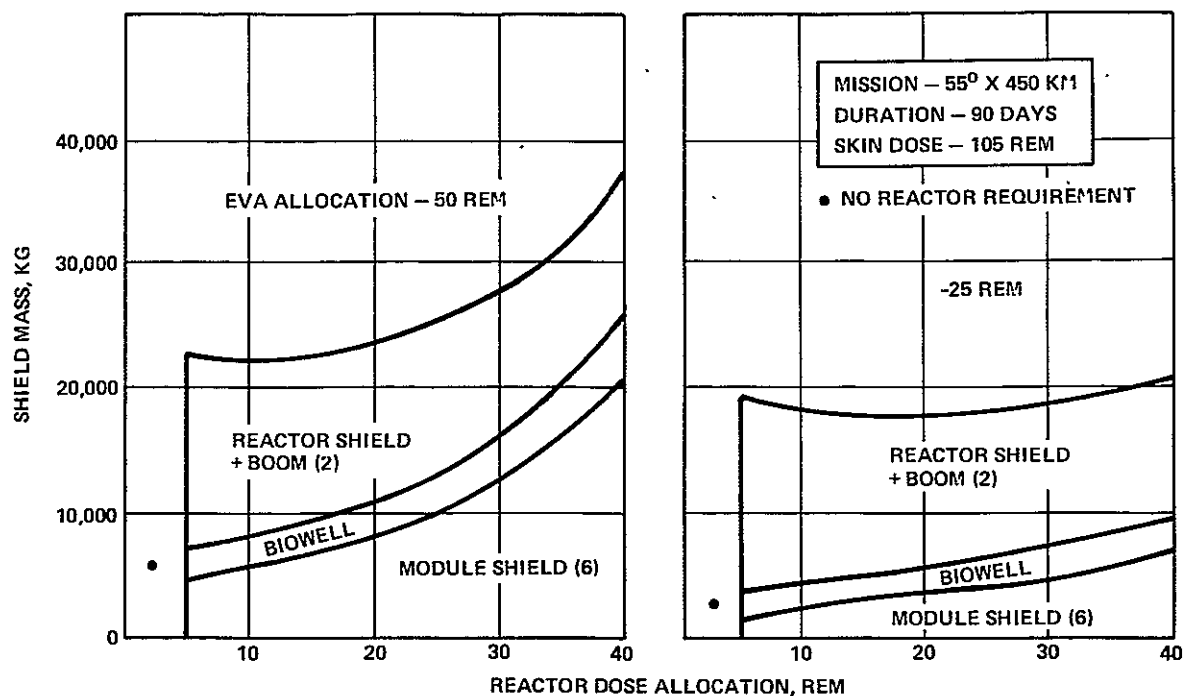


Figure 4-20. Total Shield Mass Distribution, 55 Degrees

25-rem EVA allocation. The ordinate includes the shield and boom mass for two 480-kW<sub>t</sub> reactors, a biowell, and shielding sufficient for six modules -- those needed for habitability over a 10-year program. The no-reactor case is indicated by the single data points at the left of each curve. Thus, on the left, the addition of the reactor would increase the total module and biowell shield from 6,200 to 8,000 kg, or only about 300 kg per module for the minimum-mass solution with a 50-rem EVA allocation. Thus, a small module shield penalty would allow integration of the reactor. The total shield mass data were extended to other orbit altitude missions at 55 degrees and are presented in Figure 4-21. The total shield mass increases with altitude, yet the minimum solution remains at a reactor dose allocation of 10 or 15 rem.

The conclusions reached from the radiation analysis are as follows:

A. 28.5-degree Mission

1. Nominal module design (0.1-inch-thick wall) adequate to 560 km
2. EVA can be scheduled around South Atlantic Anomaly
3. Shuttle suit adequate for EVA

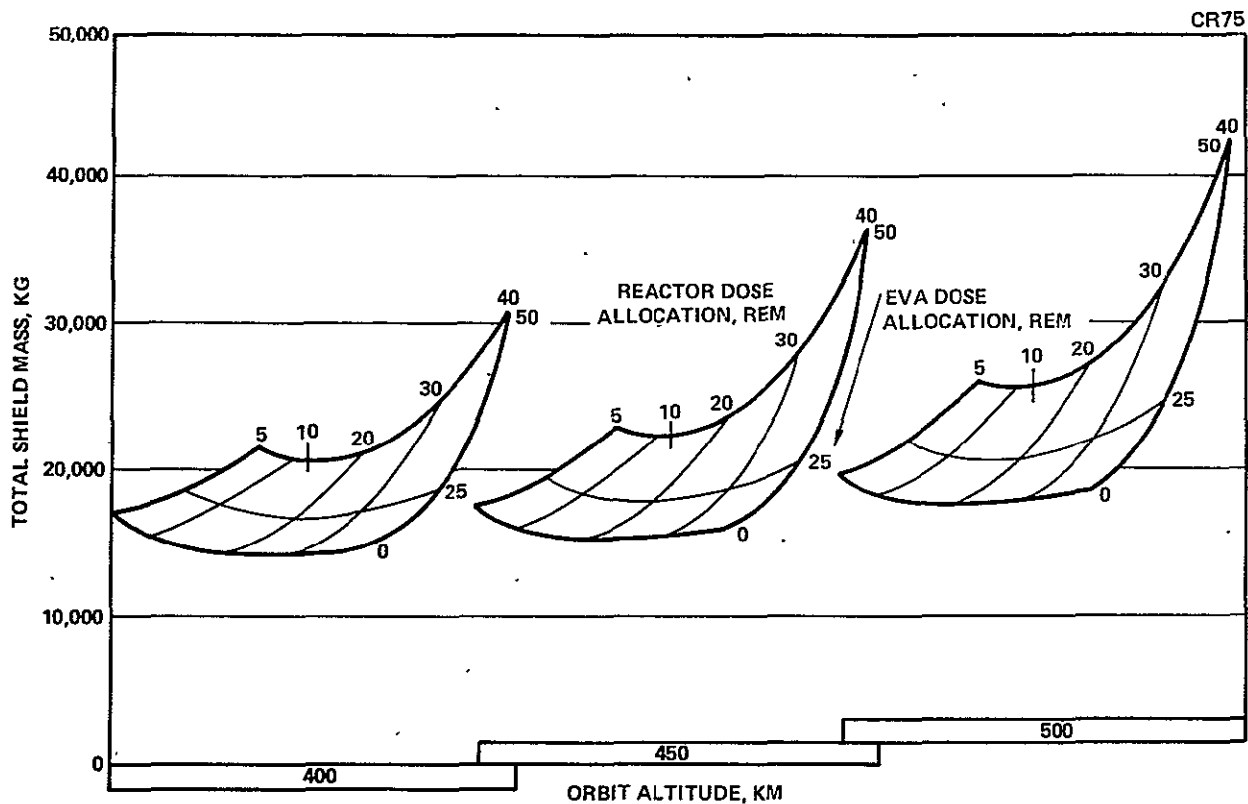


Figure 4-21. Module, Biowell, Reactor Shield Mass Required – 55 Degrees, Skin Dose, 90 Days, 480 kW<sub>t</sub>

4. Module design and EVA unaffected by reactor presence
5. BFO dose is limiting with reactor
6. Potential reactor dose allocations

|                |     |     |     |     |
|----------------|-----|-----|-----|-----|
| Altitude (km): | 400 | 450 | 500 | 550 |
| Dose (rem):    | 29  | 23  | 13  | 0   |

B. 55-degree Mission

1. Module and biowell shield needed without reactor (0.06-inch Al - 600 kg and 0.4-inch Al - 700 kg, respectively)
2. Shuttle suit provides marginal protection
3. Reactor dose can be integrated into total dose allocation
4. Skin dose is limiting
5. Space Station shield penalty due to reactor presence is 1,800 kg (300 kg for six modules)
6. Minimum mass reactor allocation equals 10 to 15 rem
7. Required reactor shield mass is 5,800 kg (12.5 rem).

#### 4.3.2 Operations Analysis

Assembly of the reactor brayton system was analyzed considering utilization of the SCB capability and then considering a Shuttle sortie capability. It was found that both approaches are feasible. Each requires unique support equipment, with the sortie mode also requiring an additional Shuttle launch.

##### Buildup

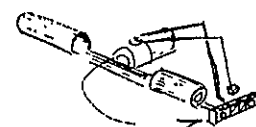
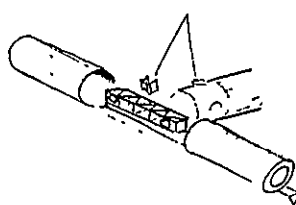
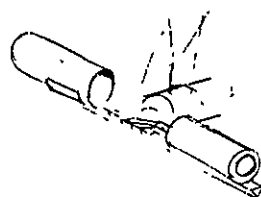
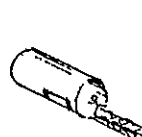
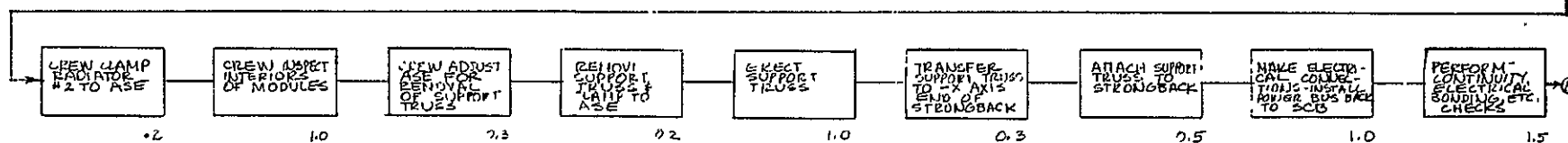
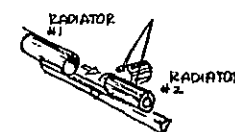
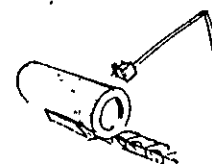
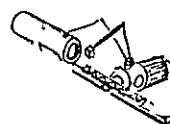
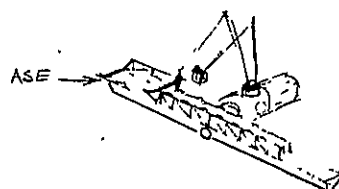
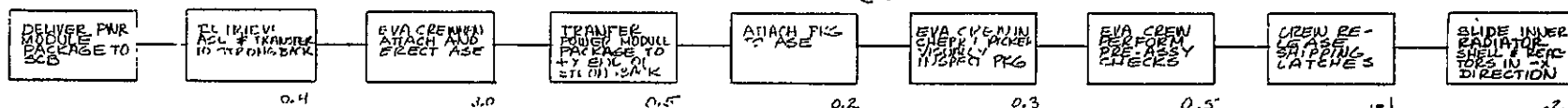
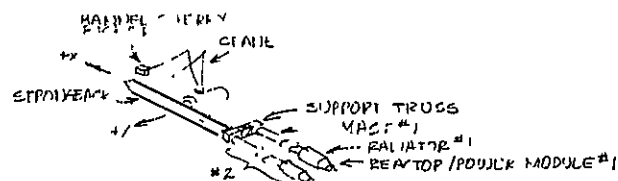
A buildup sequence using the SCB is summarized in Figure 4-22. In this buildup, both power modules are delivered in a single launch using a nested package, as described in Section 4.2. Also delivered is a special set of assembly support equipment (ASE), which is assembled and erected on the SCB strongback and provides support for the system parts as they are removed from their nested configuration and subsequently joined.

Upon completion of the ASE installation, the power module package is installed on the strongback. The outer shell, radiator No. 1, is held fixed at the end of the strongback while the remainder of the package slides out, using the ASE as a guide, and is installed at the other (-X) end of the strongback. The support truss elements are then removed from the package, erected, and installed at the -X end of the strongback. The No. 1 reactor/power/module (R/PM No. 1) is then removed from the package and installed on the end of radiator No. 1. After appropriate connections are made and equipment is checked out, the tunnel included in the interior of radiator No. 1 is partially extended to clear the end of the radiator, and the entire assembly is installed on the support truss.

The remaining package, consisting of the No. 2 system, is then transferred to the +X end of the strongback and assembled in a manner similar to that used for No. 1. It is then installed on the support truss, checked out, and the tunnel fully extended. The power module is activated, checked out, and the radiation field measured. The other power module is left in a dormant mode for later use.

The time required to accomplish this job is relatively short. As indicated in Figure 4-23, the entire operation takes only about four days, assuming a two-shift work day. Most of the work requires EVA, with the most critical

TIME IS IN HOURS



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Figure 4-22. Single Launch - Dual Power Module Build-Up (Sheet 1 of 3)

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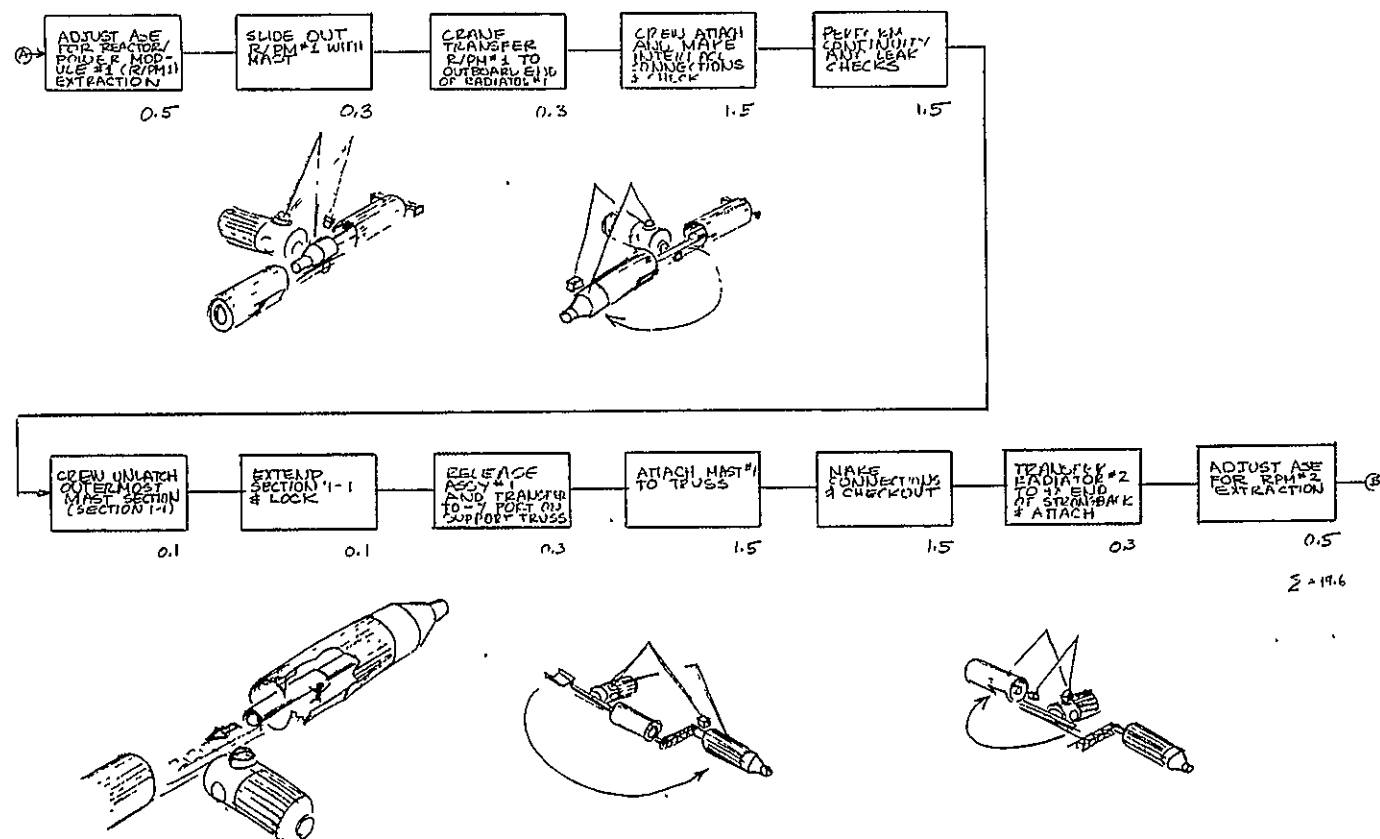
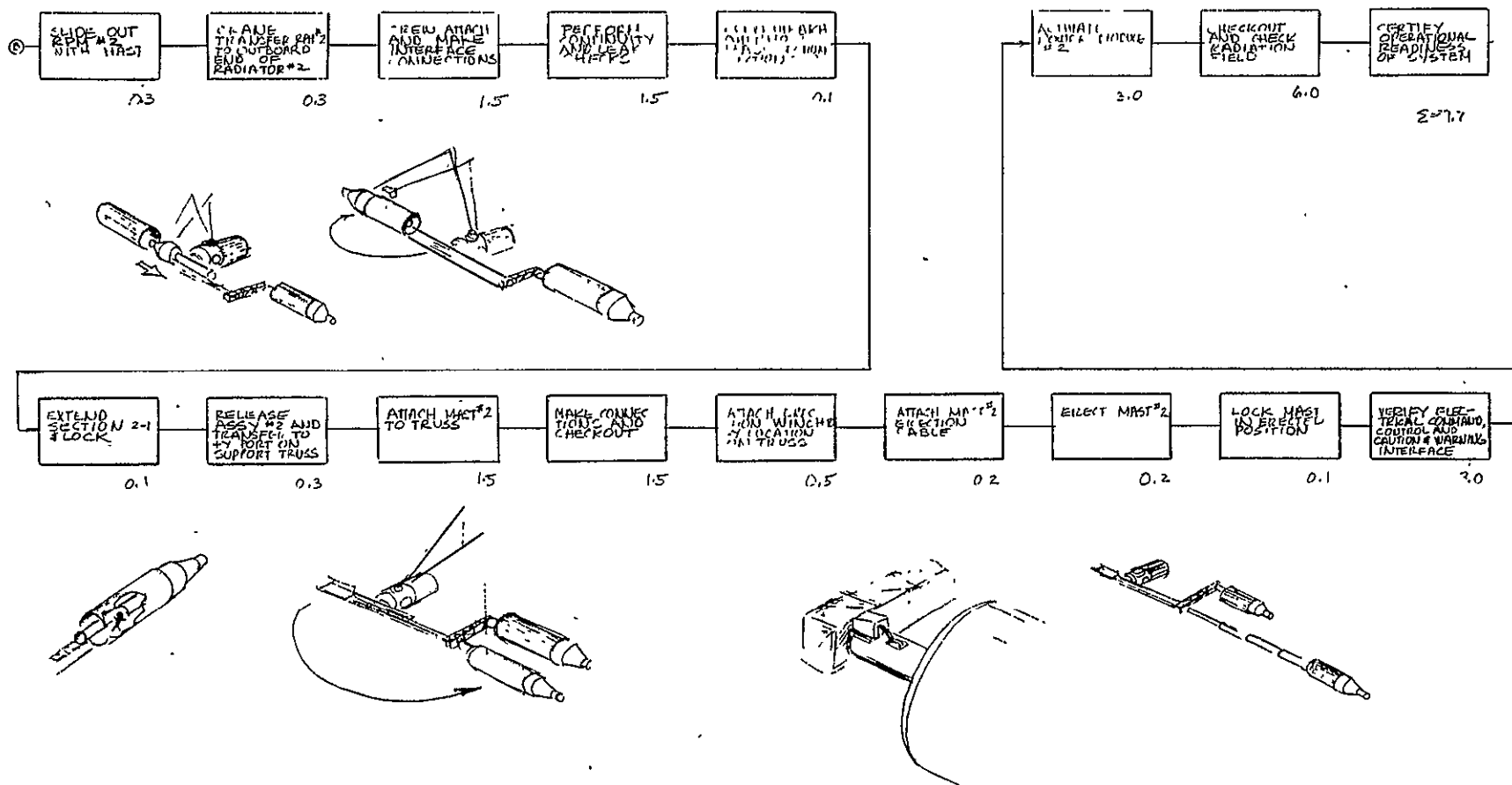


Figure 4-22. Single Launch - Dual Power Module Build-Up (Sheet 2 of 3)





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Figure 4-22. Single Launch - Dual Power Module Build-Up (Sheet 3 of 3)

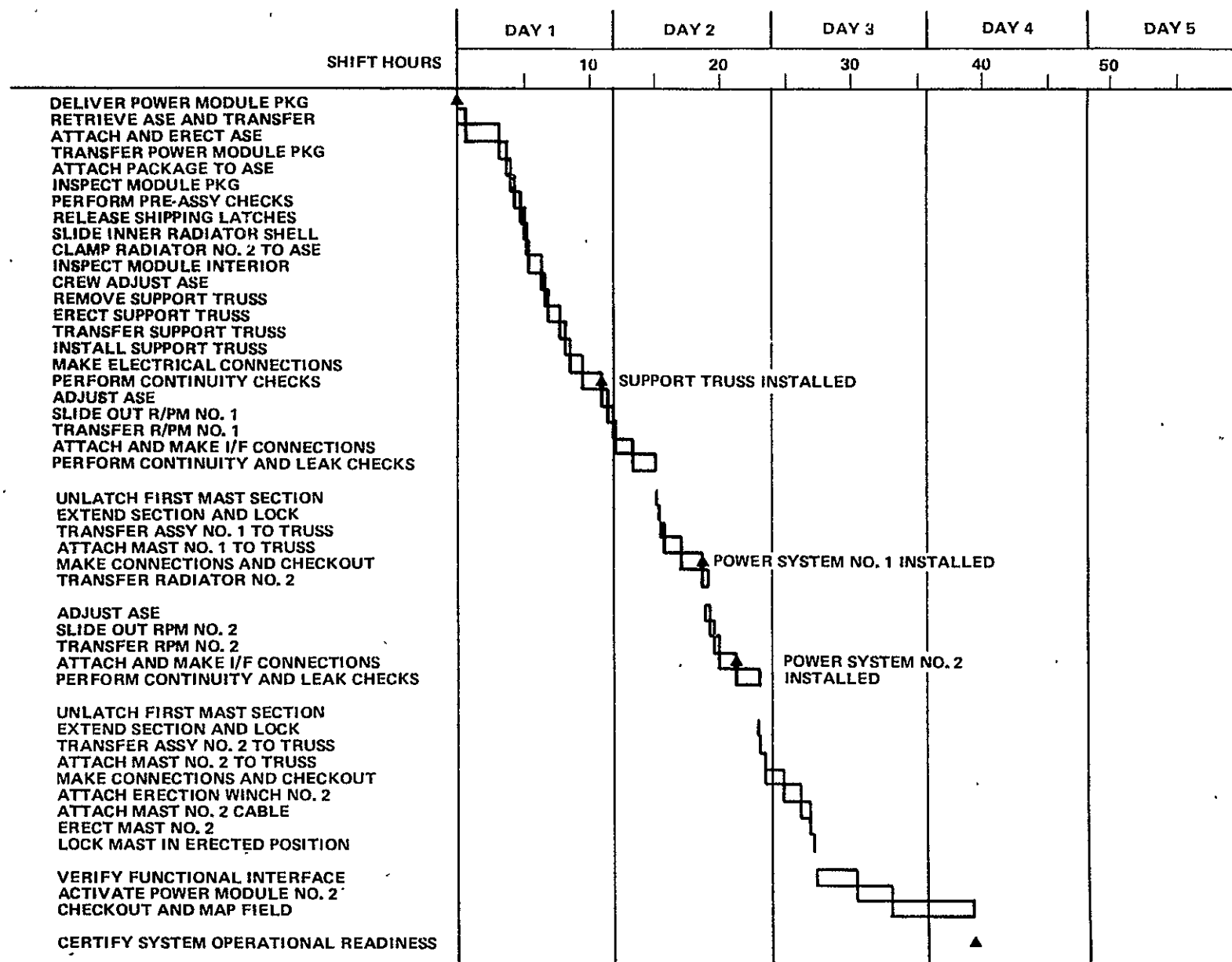


Figure 4-23. Single Launch - Dual Module Build-Up Timeline

EVA tasks being installation and checkout (particularly leak-checks) of the reactor power module to the radiators, and erection of the tunnels. Automatic erection would be very expensive, and therefore manual erection is suggested. The most sensitive task is the removal of the nested assemblies. The ASE must be properly designed and great care must be taken to avoid damage as parts with minimum clearance margins are extracted.

Sortie Mode Buildup. An optional buildup in which the first power module is assembled in a Shuttle sortie mode is indicated in Figure 4-24. Packaging and the assembly sequence are quite different in that no way was uncovered to assemble two reactor brayton systems in a Shuttle sortie mode. Accordingly, for the sortie mode, a single system is packaged for delivery (see Section 4.2). The package is delivered to orbit and installed on the Shuttle docking module, with the aid of a docking adapter which is berthed to the tunnel inside the radiator. The outer tunnel segment is then extended, which pushes the reactor/power module up through the open end of the radiator assembly. Interface connections are made, system checks performed, and the tunnel extended. The Shuttle then separates, leaving the power system in a stable mode. Because of its configuration, this system might be stabilized by using gravity gradient forces and simple dampers.

The second Orbiter then comes up, docks to the side port on the docking adapter, and berths the space construction module (SCM). A third launch brings up the strongback and support truss. A fourth launch brings up the second reactor brayton power system, which is assembled in the same way as the first system, but on the +X end of the strongback.

Comparison of Buildup Options. Assembly of the reactor brayton system is more complex on the SCB due to the fact that a nested package is used with both systems delivered simultaneously. The sortie mode is more straightforward, but only one system is delivered at a time. The alternate approach requires special ASE for handling while the sortie mode requires a docking adapter (as discussed in Section 5.3.1 for the solar brayton system assembled in a sortie mode) and a stability and control system (possibly quite

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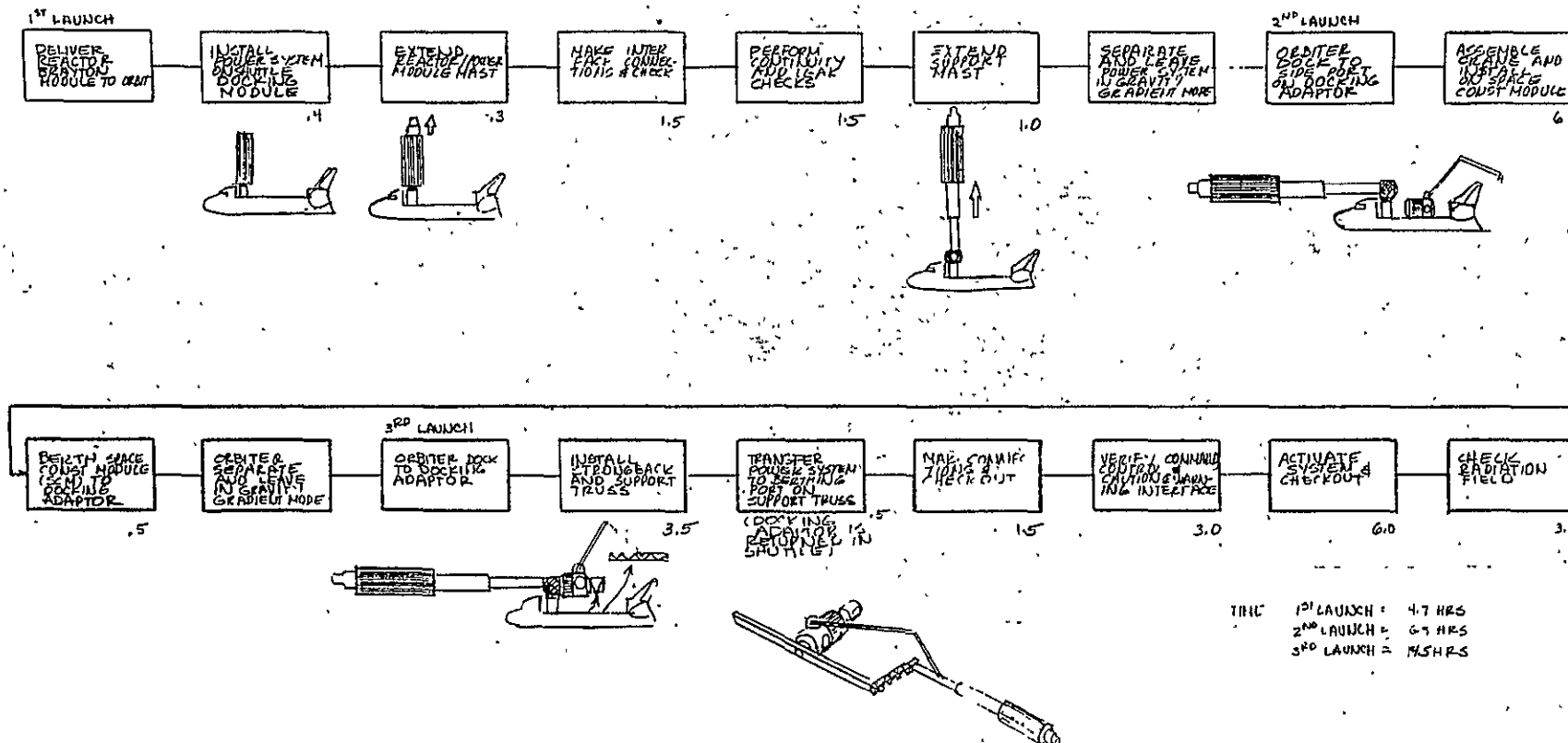


Figure 4-24. Reactor Brayton Power System Sortie Mode Delivery

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simple). In comparing the two, the complexity and cost of the nested packaging and attendant ASE must be traded against the cost of an additional Shuttle flight, incorporation of a possibly simple control system, and addition of a docking adapter. From an operational point of view, a third approach might be the most attractive; assembly of individually launched reactor brayton systems on an SCB.

The independently launched power module timeline and deployment flow are shown in Figures 4-25 and 4-26. This method was selected as the baseline for the power system delivery.

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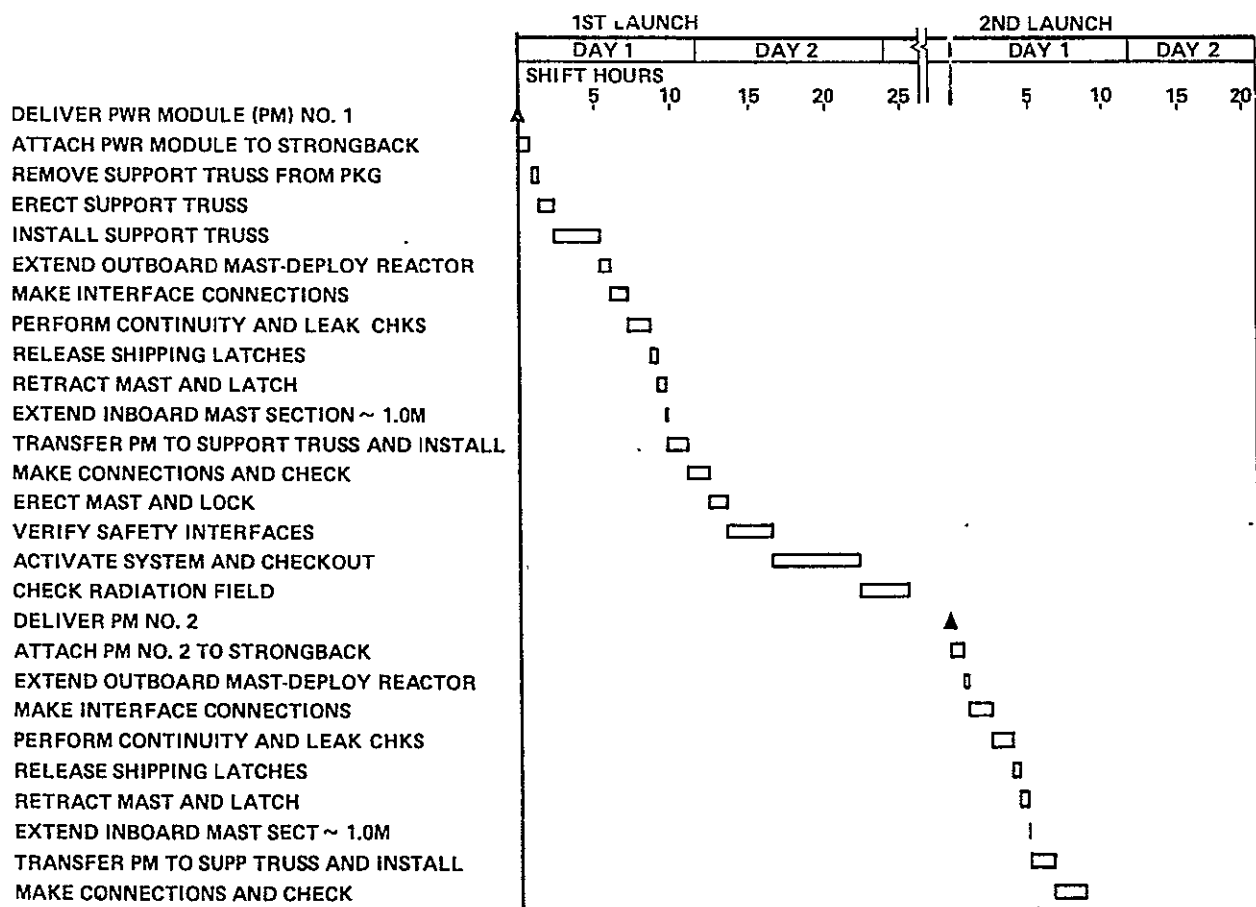


Figure 4-25. Independently Launched Reactor Brayton Power Module Timeline

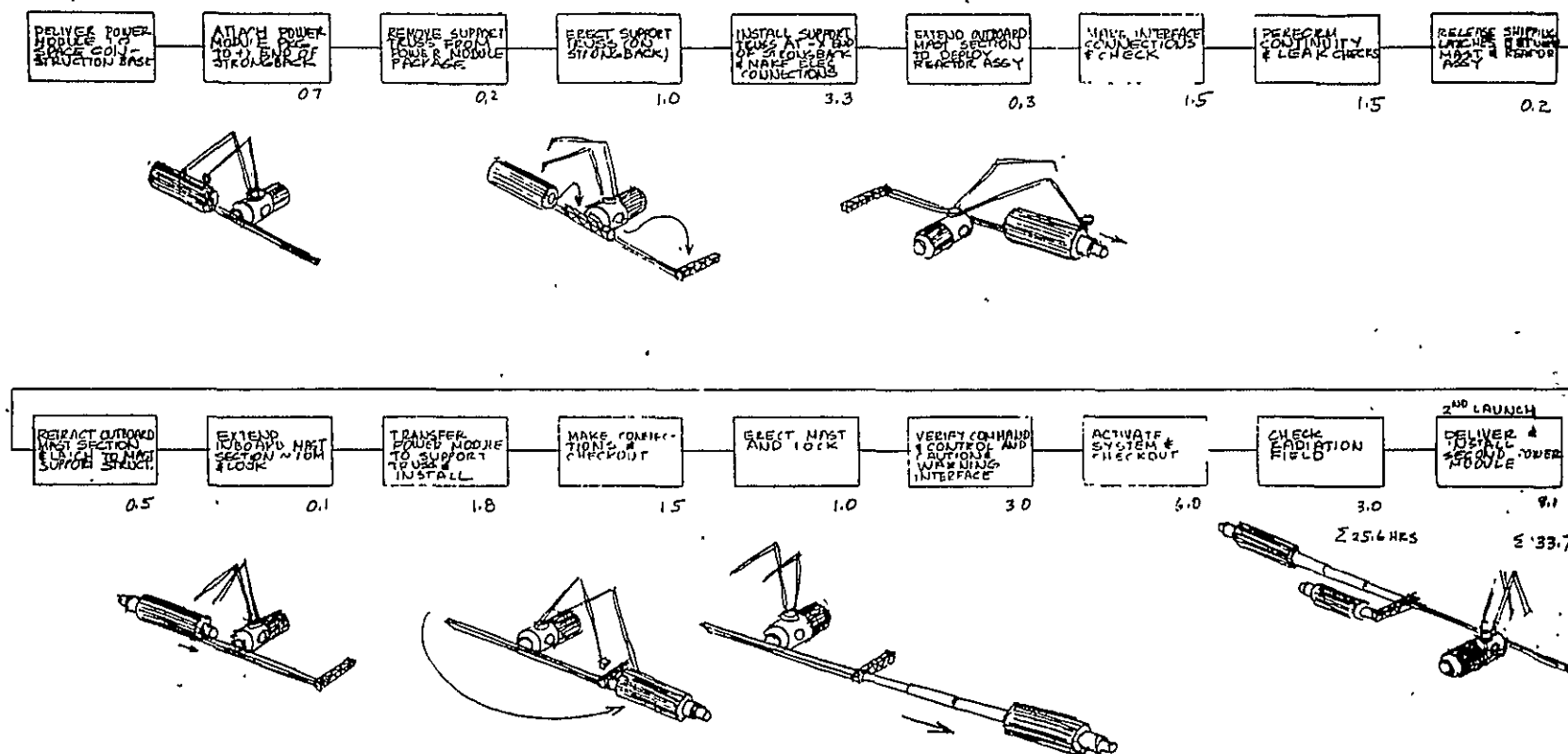


Figure 4-26. Independently Launched Power Modules (Time is in Hours)

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## Reactor Disposal

End-of-life disposal for the reactor power systems has been baselined to a high altitude where the orbit decay time will be on the order of 100 years or longer. An alternate mode of deep ocean disposal remains a possibility, with a decision to be made only after adequate safety analysis.

Disposal operations dictate low-thrust maneuvering capabilities to ensure controlled backaway, halt, and transfer to an orbit-keeping position (probably aft of the station) where final system checkout and time-phasing can occur before initiating transfer to the disposal orbit. In addition to the maneuvering capabilities, attitude control sensors and computer guidance capability will be needed, with capability for command and control either from the SCB or the ground control center. Stabilization during injection and circularization burns is also necessary. Previous analyses have selected three-axis stabilization for this operation. Further analysis will be necessary to assess the disposal options. These options are compared below.

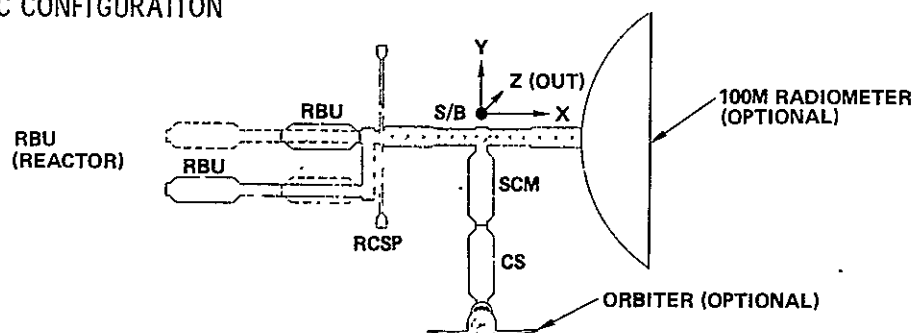
- Requirement - Place module in a long-life orbit or in specified ocean areas
- Equipment - Propulsion, stabilization, guidance, control, power
- Options
  - A. Install at beginning of life
    - 1. Always available
    - 2. Covers catastrophic SCB failure
    - 3. Avoids add-on operation
    - 4. Reliability reduction
    - 5. Radiation effects
    - 6. Needs periodic verification
    - 7. Maintenance requires shutdown of reactor system
  - B. Install at end of life
    - 1. Not always available
    - 2. Requires shielding and cooldown for add-on operation
    - 3. Requires stable module for docking
    - 4. Allows late ground checkout
    - 5. Easy maintenance.

#### 4.3.3 Guidance and Control Analysis

The guidance and control analysis for the reactor brayton system considered two reactor brayton units at one end of the strongback, as shown in Figure 4-27. Only one unit is extended at a time and there are therefore two configuration combinations that were studied (marked in the figure by "solid" or "dotted" lines). The study considered the presence or absence of the 100-meter radiometer at the other end of the strongback, representative of one of the largest mission hardware elements considered on the Space Station Study program. The analysis allowed consideration of the effect of the extremes in size, mass, and aerodynamic forces on stability, control, and orbit-keeping of the vehicle. Additionally, the study considered the presence or absence of the Orbiter. Although the Orbiter as a logistics vehicle will nominally be attached only for small percentages of the time, this was considered for the sake of completeness and to give additional insight into the effect of its large mass and displacement of the center of gravity from the docking axis.

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##### ● BASIC CONFIGURATION



##### ● CONFIGURATIONS STUDIED (WITHOUT & WITH ORBITER)

- BASIC ("SOLID" REACTORS)
- BASIC ("SOLID" REACTORS) + 100-M RADIOMETER
- BASIC ("DOTTED" REACTORS)
- BASIC ("DOTTED" REACTORS) + 100-M RADIOMETER
- COMBINATIONS OF THE ABOVE WITH ONE OR BOTH RBU'S ROTATED FROM POSITION TO CHANGE PRINCIPAL AXIS TILT

Figure 4-27. Reactor Brayton Attitude Control Analysis



Orientation requirements of the orbiting vehicle depend upon accommodation of the mission hardware elements and the desirability to reduce the cost and resources necessary to maintain the vehicle in orbit. It has been established that any element requiring highly accurate pointing will contain the instrumentation and actuation means to satisfy that requirement. The SCB would provide the coarse pointing and stabilize the entire vehicle against the environmental torques. If the mission hardware element requires no particular orientation, then an orientation requiring minimum resources is permitted.

As discussed in Section 5.3.2, a stable SCB orientation exists relative to the vertical and orbit plane. This will generally be the preferred long-term, low-resource orientation unless the associated drag necessitates an excessively large amount of propellant for orbit-keeping. The characteristics of the principal axes for the stable gravity gradient orientation are as follows:

- A. Roll. The roll principal axis is directed along the velocity vector. The value of the moment of inertia about this axis,  $I_R$ , is intermediate to the values about the two other principal axes.
- B. Pitch. The pitch principal axis is directed perpendicular to the orbit plane. The value of the moment of inertia about this axis,  $I_P$ , is the largest of the three principal axes.
- C. Yaw. The yaw principal axis is directed along the vertical. The value of the moment of inertia about this axis,  $I_Y$ , is the smallest of the three principal axes.

Orientation other than this will require the expenditure of resources (such as attitude control propellant) to hold orientation against gravity gradient torques. These orientations will come from particular test, calibration, or data-gathering operations with mission hardware items.

#### Analysis

Analysis of the moments of inertia of the reactor brayton configuration of Figure 4-27 is summarized in Tables 4-7 through 4-10. The first column is an indication of the rotation of the principal axes from the geometrical

Table 4-7

CONFIGURATION: REACTOR BRAYTON  
 REACTOR ALONG STRONGBACK RETRACTED, OTHER REACTOR EXTENDED

| Axis            | Rotation<br>(deg) | $I \times 10^{-6}$<br>(Slug-ft <sup>2</sup> ) | $\Delta I \times 10^{-6}$<br>(Slug-ft <sup>2</sup> ) | $T_{MAX}^*$<br>(ft-lb) | $T/\theta^*$<br>(ft-lb/deg) | $\dot{\theta}_{90}^*$<br>(deg/sec) | $P_P$<br>(min) | $P_Y$<br>(min) | $P_R$<br>(min) |
|-----------------|-------------------|-----------------------------------------------|------------------------------------------------------|------------------------|-----------------------------|------------------------------------|----------------|----------------|----------------|
| Without Orbiter |                   |                                               |                                                      |                        |                             |                                    |                |                |                |
| X <sub>P</sub>  | 10.7              | 4.307                                         | 3.83                                                 | 7.21                   | 0.25                        | 0.105                              | 57.3           | 99.2           | 49.6           |
| Y <sub>P</sub>  | 13.8              | 86.353                                        | 85.88                                                | 161.6                  | 5.64                        | 0.111                              | 54.1           | 93.8           | 46.9           |
| Z <sub>P</sub>  | 9.3               | 90.183                                        | 82.05                                                | 154.4                  | 5.39                        | 0.106                              | 56.6           | 98.0           | 49.0           |

Conclusions: GG stable orientation is: X<sub>P</sub> - VERT, Y<sub>P</sub> - AVV, Z<sub>P</sub> - POP

|                |      |         |        |       |      |       |      |       |      |
|----------------|------|---------|--------|-------|------|-------|------|-------|------|
| With Orbiter   |      |         |        |       |      |       |      |       |      |
| X <sub>P</sub> | 24.3 | 26.669  | 24.75  | 46.56 | 1.63 | 0.107 | 56.0 | 97.1  | 48.5 |
| Y <sub>P</sub> | 34.9 | 119.533 | 117.62 | 221.3 | 7.72 | 0.110 | 54.4 | 94.3  | 47.1 |
| Z <sub>P</sub> | 24.7 | 144.284 | 92.86  | 174.7 | 6.10 | 0.098 | 61.3 | 106.1 | 53.0 |

Conclusions: GG stable orientation is: X<sub>P</sub> - VERT, Y<sub>P</sub> - AVV, Z<sub>P</sub> - POP

\*Given as in-plane (pitch) characteristics

Note: orbit altitude = 450 km

AVV = Along Velocity Vector

POP = Perpendicular to Orbit Plane

VERT = Along the Vertical

Table 4-8

CONFIGURATION: REACTOR BRAYTON  
 REACTOR ALONG STRONGBACK EXTENDED, OTHER REACTOR RETRACTED

| Axis            | Rotation<br>(deg) | $I \times 10^{-6}$<br>(Slug-ft <sup>2</sup> ) | $\Delta I \times 10^{-6}$<br>(Slug-ft <sup>2</sup> ) | $T_{MAX}^*$<br>(ft-lb) | $T/\theta^*$<br>(ft-lb/deg) | $\dot{\theta}_{90}^*$<br>(deg/sec) | $P_P$<br>(min) | $P_Y$<br>(min) | $P_R$<br>(min) |
|-----------------|-------------------|-----------------------------------------------|------------------------------------------------------|------------------------|-----------------------------|------------------------------------|----------------|----------------|----------------|
| Without Orbiter |                   |                                               |                                                      |                        |                             |                                    |                |                |                |
| $X_P$           | 6.6               | 6.096                                         | 5.50                                                 | 10.35                  | 0.36                        | 0.106                              | 56.5           | 98.4           | 49.2           |
| $Y_P$           | 11.0              | 84.622                                        | 84.03                                                | 158.1                  | 5.52                        | 0.111                              | 54.2           | 93.8           | 46.9           |
| $Z_P$           | 4.9               | 90.125                                        | 78.53                                                | 147.7                  | 5.16                        | 0.104                              | 57.8           | 100.2          | 50.1           |

Conclusions: GG stable orientation is:  $X_P$  - VERT,  $Y_P$  - AVV,  $Z_P$  - POP

|              |      |         |        |       |      |       |      |       |      |
|--------------|------|---------|--------|-------|------|-------|------|-------|------|
| With Orbiter |      |         |        |       |      |       |      |       |      |
| $X_P$        | 21.7 | 30.917  | 28.03  | 52.73 | 1.84 | 0.106 | 56.7 | 98.2  | 49.1 |
| $Y_P$        | 31.3 | 115.943 | 113.06 | 212.7 | 7.42 | 0.110 | 54.7 | 94.7  | 47.3 |
| $Z_P$        | 22.4 | 143.974 | 85.03  | 160.0 | 5.58 | 0.085 | 70.3 | 121.7 | 60.8 |

Conclusions: GG stable orientation is:  $X_P$  - VERT,  $Y_P$  - AVV,  $Z_P$  - POP

\*Given as in-plane (pitch) characteristics:

Note: Orbit altitude = 450 km

AVV = Along Velocity Vector

POP = Perpendicular to Orbit Plane

VERT = Along the Vertical

Table 4-9

CONFIGURATION: REACTOR BRAYTON + 100M RADIOMETER  
 REACTOR ALONG STRONGBACK RETRACTED, OTHER REACTOR EXTENDED

| Axis            | Rotation<br>(deg) | $I \times 10^{-6}$<br>(Slug-ft <sup>2</sup> ) | $\Delta I \times 10^{-6}$<br>(Slug-ft <sup>2</sup> ) | $T_{MAX}^*$<br>(ft-lb) | $T/\theta^*$<br>(ft-lb/deg) | $\dot{\theta}_{90}^*$<br>(deg/sec) | $P_P$<br>(min) | $P_Y$<br>(min) | $P_R$<br>(min) |
|-----------------|-------------------|-----------------------------------------------|------------------------------------------------------|------------------------|-----------------------------|------------------------------------|----------------|----------------|----------------|
| Without Orbiter |                   |                                               |                                                      |                        |                             |                                    |                |                |                |
| $X_P$           | 12.2              | 37.001                                        | 9.13                                                 | 17.18                  | 0.60                        | 0.055                              | 108.7          | 188.2          | 94.1           |
| $Y_P$           | 29.2              | 243.196                                       | 197.06                                               | 370.7                  | 12.94                       | 0.100                              | 60.0           | 103.9          | 51.9           |
| $Z_P$           | 30.2              | 234.063                                       | 206.20                                               | 387.9                  | 13.54                       | 0.104                              | 57.5           | 99.6           | 49.8           |

Conclusions: GG stable orientation is:  $X_P$  - VERT,  $Y_P$  - POP,  $Z_P$  - AVV

With Orbiter

|       |      |         |        |       |       |       |      |       |      |
|-------|------|---------|--------|-------|-------|-------|------|-------|------|
| $X_P$ | 12.3 | 83.013  | 39.21  | 73.76 | 2.57  | 0.076 | 78.6 | 136.1 | 65.0 |
| $Y_P$ | 4.9  | 247.938 | 204.13 | 384.0 | 13.41 | 0.101 | 59.5 | 103.1 | 51.5 |
| $Z_P$ | 11.3 | 287.143 | 164.93 | 310.3 | 10.83 | 0.094 | 71.2 | 123.4 | 61.7 |

Conclusions: GG stable orientation is:  $X_P$  - VERT,  $Y_P$ , AVV,  $Z_P$  - POP

\*Given as in-plane (pitch) characteristics

Note: orbit altitude = 450 km

AVV = Along Velocity Vector

POP = Perpendicular to Orbit Plate

VERT = Along the Vertical

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Table 4-10

CONFIGURATION: REACTOR BRAYTON + 100M RADIOMETER  
 REACTOR ALONG STRONGBACK EXTENDED, OTHER REACTOR RETRACTED

| Axis            | Rotation<br>(deg) | $I \times 10^{-6}$<br>(Slug-ft <sup>2</sup> ) | $\Delta I \times 10^{-6}$<br>(Slug-ft <sup>2</sup> ) | $T_{MAX}^*$<br>(ft-lb) | $T/\theta^*$<br>(ft-lb/deg) | $\dot{\theta}_{90}^*$<br>(deg/sec) | $P_P$<br>(min) | $P_Y$<br>(min) | $P_R$<br>(min) |
|-----------------|-------------------|-----------------------------------------------|------------------------------------------------------|------------------------|-----------------------------|------------------------------------|----------------|----------------|----------------|
| Without Orbiter |                   |                                               |                                                      |                        |                             |                                    |                |                |                |
| X <sub>P</sub>  | 11.6              | 36.422                                        | 7.63                                                 | 14.36                  | 0.50                        | 0.051                              | 117.9          | 204.3          | 102.1          |
| Y <sub>P</sub>  | 22.4              | 226.508                                       | 182.45                                               | 343.2                  | 11.98                       | 0.100                              | 60.2           | 104.2          | 52.1           |
| Z <sub>P</sub>  | 24.9              | 218.876                                       | 190.09                                               | 357.6                  | 12.48                       | 0.104                              | 57.9           | 100.3          | 50.2           |

Conclusions: GG stable orientation is: X<sub>P</sub> - VERT, Y<sub>P</sub> - POP, Z<sub>P</sub> - AVV

|                |      |         |        |       |       |       |      |       |      |
|----------------|------|---------|--------|-------|-------|-------|------|-------|------|
| With Orbiter   |      |         |        |       |       |       |      |       |      |
| X <sub>P</sub> | 11.1 | 78.623  | 35.02  | 65.89 | 2.30  | 0.074 | 80.9 | 140.1 | 70.1 |
| Y <sub>P</sub> | 2.7  | 232.439 | 188.84 | 355.3 | 12.40 | 0.100 | 59.9 | 103.7 | 51.9 |
| Z <sub>P</sub> | 11.4 | 267.462 | 153.82 | 289.4 | 10.10 | 0.084 | 71.2 | 123.3 | 61.7 |

Conclusions: GG stable orientation is: X<sub>P</sub> - VERT, Y<sub>P</sub> - AVV, Z<sub>P</sub> - POP

\*Given as in-plane (pitch) characteristics  
 Note: Orbit altitude = 450 km  
 AVV = Along Velocity Vector  
 POP = Perpendicular to Orbit Plate  
 VERT = Along the Vertical

axes. The numbers given are the magnitude of the rotation from the indicated axis, but do not include the full  $3 \times 3$  matrix of inverse direction cosines. The I and  $\Delta I$  columns of the tables are the moment of inertia parameters of the configuration upon which the rest of the columns are based. Because yaw and roll in orbit coordinates are coupled through gyroscopic torques, the next three columns are presented as if each of the three axes is an uncoupled pitch (i. e., is perpendicular to the orbit plane). They represent the maximum torque per axis (at 45 degree orientation of the axis from the vertical), the torque per degree from the null point, and the "capture" velocity (i. e., the velocity below which the vehicle will oscillate about the stable point). Low capture velocities require a tight attitude control before the vehicle is released to stable-free oscillation. The last three columns represent the period of oscillation for small perturbations, assuming that each of the axes is stable as a pitch, yaw, or roll axis. The numbers enclosed represent the applicable values assuming complete, three-axis gravity gradient stabilization.

Tables 4-7 and 4-8 summarize the analysis for the case with no mission hardware included.

For the cases shown, the  $Z_p$ -axis (Figure 4-27) is the pitch (POP) axis, the  $Y_p$ -axis is the roll (AVV) axis, and the  $X_p$ -axis is the yaw (VERT) axis. This condition holds with the Orbiter present; however, the Orbiter increases the torque sensitivities by approximately 40 percent and makes a significant increase in the tilt of the principal axis alignment.

The addition of a large mission hardware element such as a 100-meter radiometer to the vehicle increases the principal axis tilt and changes the desired orientation without the Orbiter, as indicated in Tables 4-9 and 4-10. The  $X_p$ -axis is still vertical, but the other two axes are interchanged. With the Orbiter present, the axis tilt decreases, and the preferred orientation about the vertical  $X_p$ -axis differs, depending upon which reactor is activated.

The reactor brayton configurations present consistently strong stable orientations to the vertical and orbit plane. Because of the large masses associated with the reactor brayton units, the gradient torques are two-to-four times larger than those of the solar brayton units. Thus, to hold the vehicle at a

fixed orientation (that is, fixed relative to inertial space or orbit coordinates), the reactor brayton system will require two to four times as much propellant or two to four times as much angular momentum storage over a given period of time. This effect cannot be evaluated until fixed orientation requirements are established for the program.

The large reactors with their freedom of orientation raises a question about whether articulating the booms can change the principal axis tilt so that a change in SCB stable orientation can be effected. In this sense they can become trim devices. The options are to swing the extended reactor out-board for either reactor arrangement or to swing both reactors in or out of the plane of the paper (Figure 4-27). The axis tilts associated with articulating the reactors 45 degrees vary from 15 to 20 degrees for the configurations without the Orbiter and without the 100-meter radiometer. At articulations of 90 degrees, principal axis exchanges occur and there seems to be little benefit. Configurations with the 100-meter radiometer and/or the Orbiter give axis interchanges and poorly defined results at an articulation angle of 45 degrees. Thus, using articulation of the reactors as a trimming concept appears feasible under certain conditions, but limited in application to all configurations.

The effect of aerodynamic forces without mission hardware on attitude control is rather minor with this configuration. The steady aerodynamic torque, assuming solar maximum conditions,  $S_{10.7} = 170$ , will offset the attitude by 0.14 degree from the gravity gradient trim condition. Due to the cyclic influence of the diurnal bulge, this trim point will vary from 0.09 to 0.19 degrees. When the 100-meter radiometer is added to the configuration, the trim angle will increase to  $1 \pm 0.3$  degrees. The addition of the Orbiter to these configurations will have a small effect.

Orbit-keeping and attitude control propellant were determined for the configuration with and without the 100-meter radiometer. The data associated with orbit maintenance requirements are presented in Table 4-11. As indicated, the radiometer has a large effect on all factors involved. The large drag area dominates the orbit-keeping requirements and influences the projected lifetime and orbit-keeping scheduling. Scheduled orbit-keeping

Table 4-11  
REACTOR BRAYTON ORBIT-KEEPING REQUIREMENTS  
450 km - NO ORBITER

|                                            | No Mission<br>Hardware | With 100-m<br>Radiometer |
|--------------------------------------------|------------------------|--------------------------|
| CdA (ft <sup>2</sup> )                     | 8,430                  | 75,600                   |
| W/CdA (lb/ft <sup>2</sup> )                | 26.1                   | 4.4                      |
| Lifetime**<br>(weeks)                      | 92                     | 16                       |
| Orbit-keeping***<br>interval (days)        | 51                     | 8.5                      |
| Drag Propellant***<br>(kg/mo)              | 69                     | 622                      |
| Attitude Control*<br>Propellant (kg/mo)*** | 108                    | 258                      |
| Total (kg/mo)                              | 177                    | 880                      |

\*No CMG's

\*\*Assuming no orbit-keeping

\*\*\*5% interval factor penalty,  $I_{sp} = 270$  sec

permits a decay in the constant orbit altitude, relative to continuous thrust operation, and incurs a 5 percent propellant penalty. Although the vehicle is assumed to be gravity gradient stable (slightly modified by aerodynamic torques), an allowance for attitude control propellant was made as a function of moment of inertia per axis. Although this is a judgmental factor, the values represent an allowance for capture maneuvers, reorientation, etc. It appears that CMG's are not required as part of the attitude control system unless mission hardware warrants them.

### Summary

A summary is presented of the attitude control and flight impacts of the reactor brayton system. The attitude control summary is given below:

- Gravity gradient stabilization is recommended for long-term orientation.
- Control moment gyros (CMG's) are recommended if orientation with zero gravity gradient moment, but unstable slope, is used.



- Stable orientation (no Orbiter)

| Basic        | Basic + 100-M Radiometer |
|--------------|--------------------------|
| $X_P$ - VERT | $X_P$ - VERT (weak)      |
| $Y_P$ - AVV  | $Y_P$ - POP              |
| $Z_P$ - POP  | $Z_P$ - AVV              |

- No software required for power system pointing.
- Gravity gradient torques off trim 2 to 4 times more than solar brayton.
- Orbit-keeping propellant less than solar brayton, but total propellant requirements are similar.
- Aerodynamic trim offset is smaller than solar brayton.
- Reactor brayton units can be articulated to change principal/geometric axis alignment as much as 20 degrees.

Comparison of this system with the solar brayton system is limited to the "trimmed" orientation for each power module - SCB configuration.

#### 4.3.4 Thermal Analysis

Thermal analysis addressed the design and performance of equipment necessary to transport source heat from the reactor to the brayton cycle working fluid and to remove heat and reject it to space at the system heat sink. In this area, a preliminary design was prepared for thermal components where available data were lacking or inadequate. Supporting analyses were performed to verify performance and develop the information necessary for the preliminary design.

The thermal requirements, analytical approach, and study results are discussed in the following text.

This section describes the effort performed to analyze and generate a preliminary design for a heat pipe radiator. The analysis provided basic design parameters and verified the performance of the resulting concept.

Previous brayton cycle space radiator systems have utilized configurations composed of multiple tubes with a pumped heat transfer fluid system for

heat rejection. The current investigation considers the use of heat pipes to replace the heat transfer fluid tubes on the radiator skin. These heat pipes would be coupled with the brayton cycle fluid cooler by an intermediate pumped fluid loop. Heat pipes offer the following advantages: (1) reduction in the total surface area of the pumped fluid system exposed to potential meteorite penetration, and (2) loss of a single heat pipe would not significantly affect the radiator performance, whereas the penetration of one of the cooling tubes of the pumped systems would result in complete loss of radiator capability.

The result was a preliminary radiator design which uses 150 heat pipes with mercury as the working fluid. Each heat pipe has a 180-degree semi-circular shape. Two rows of the heat pipes run along opposite sides of the cylindrical radiator. Redundant distribution loops transport the heat from the coolers (heat sink heat exchangers) to the evaporative section of the heat pipes where heat is transferred to the radiator.

The weight of the radiator and heat sink distribution system is 1,648 kg (3634 lb) for a 200 m<sup>2</sup> design. This design was analyzed at beta angles of 0, 78, and 90 degrees, and the design possesses a performance margin of at least 37 percent. This design margin can be used for the secondary radiator area requirements which are not yet defined.

The heat source heat exchanger concept was scaled from the unit sized for the solar brayton power system described in Section 5.3.3. This scaled unit weighs about 652 kg (1,434 lb). The scaling was done to provide weight estimates for the missing system element.

#### Requirements

Key quantitative requirements for the thermal components in the reactor brayton power system are given in Table 4-12. These requirements must be satisfied for the total system to operate at the original design point conditions.

Table 4-12  
THERMAL EQUIPMENT REQUIREMENTS FOR 120-kWe  
REACTOR BRAYTON POWER SYSTEM

| Component/Item             | Requirement        |
|----------------------------|--------------------|
| Heat Source Heat Exchanger |                    |
| Heat transferred           | 480 kW             |
| High-temperature inlet     | 1,492 K (2,226° F) |
| Low-temperature inlet      | 992 K (1,326° F)   |
| Low-temperature outlet     | 1,273 K (1,831° F) |
| Working fluid              | He/Xe              |
| Heat Sink Heat Exchanger   |                    |
| Heat transferred           | 360 kW             |
| High-temperature inlet     | 633 K (679° F)     |
| High-temperature outlet    | 428 K (310° F)     |
| Low-temperature inlet      | 608 K (634° F)     |
| Low-temperature outlet     | 402 K (264° F)     |
| Radiator                   |                    |
| Heat rejected              | 360 kW             |
| Inlet fluid temperature    | 608 K (634° F)     |
| Outlet fluid temperature   | 402 K (264° F)     |
| Radiator orientation       | Gravity gradient   |
| Inclination                | 0 to 90°           |

Functional requirements for heat sink heat exchange are to transfer heat from the He/Xe working fluid to the radiator distribution fluid. The radiator consists of heat pipes attached to a cylindrical shield/fin shell. Heat from the radiator distribution fluid is transferred to the evaporative area of the heat pipes where it is then distributed to the radiator surface for rejection to the surroundings.

The heat source heat exchanger receives heat from the reactor via heat pipes and transfers the heat to the brayton cycle working fluid.

### Analysis

This section describes the assumptions, approach, and results of the analyses.

Radiator. A heat pipe radiator was assumed for the rejection of heat sink heat. The configuration analyzed is shown in Figure 4-28. A Dowtherm A

CR75

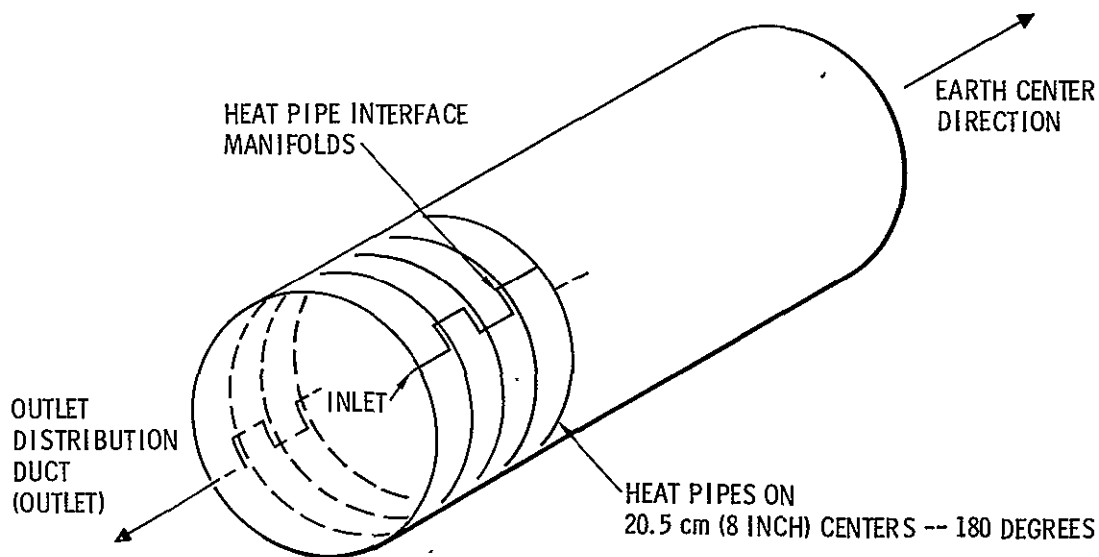


Figure 4-28. Reactor Brayton Radiator Configuration

distribution duct runs down the length of the radiator and returns along the opposite side. (Redundant distribution ducts and pumps are assumed.) Heat pipes using mercury as a working fluid are located on 20.5-cm (8 inch) centers around the periphery of the radiators. Each heat pipe runs half way around the radiators and each is located so that the distribution ducts intersect at the center of the heat pipes. Two sets of heat pipes are located on the surface, one on each side of the radiator.

The outside radiator surface acts as a finned surface. It is constructed of 0.5-mm (0.02 inch) aluminum and the heat pipes are bonded to this surface with an agent which has good heat transfer performance. The heat pipes are made of stainless steel to prevent corrosion by the heat pipe fluid (mercury). Stainless steel is also used for the distribution ducts to prevent corrosion due to dissimilar materials.

Analytical effort on the radiator consisted of sizing analysis and heat rejection capability calculations.

Heat Rejection. The thermal model used was an equivalent cylinder consisting of 12 flat plates so that a simplified closed-form analysis could be used. An average radiator temperature was used, which further simplified the task. Heat rejection capability was calculated for each surface and then summed to obtain the total capability. This was done for 12 points around the orbit, in 30-degree increments. Heat influx was considered from earth IR, earth albedo, and direct solar sources.

Calculations were performed for a thermal coating which was relatively degraded,  $\alpha_s/\epsilon_t = 0.3$ . Also; three separate beta angles were considered equal to 0, 78, and 90 degrees. The last two result in the vehicle being in the sun during the total orbit.

The sink temperature approach was used wherein:

$$T_s = \left\{ \frac{1}{\sigma} \left[ \frac{\alpha_s}{\epsilon_t} SF_s + \frac{\alpha_s}{\epsilon_t} a SF_{E(SR)} + \frac{(1-a)}{4} SF_{ET} \right] \right\}^{1/4}$$

where:

- $T_s$  = sink temperature
- $\alpha_s$  = absorptivity in solar wavelength
- $S$  = solar constant
- $F_s$  = solar view factor
- $a$  = earth albedo
- $F_{E(SR)}$  = solar reflected view factor
- $F_{ET}$  = earth IR (thermal) view factor
- $\sigma$  = Stefan Boltzmann's Constant

The sink temperature was calculated for each radiator flat equivalent surface and position in orbit and was used to calculate the heat rejection as follows:

$$\frac{Q}{A} = \sigma \epsilon_t (T_R^4 - T_s^4)$$

where:

- $\frac{Q}{A}$  = heat rejected per unit area
- $T_R$  = average radiator surface temperature

Results of the analyses are shown in Figure 4-29 which gives  $Q/A$  as a function of average radiator temperature; the average between inlet and outlet radiator distribution fluid temperature is 505 K (449° F). A typical average radiator surface temperature is 475 K (395° F) which allows a 30 K temperature drop from distribution fluid to finned surface. The temperature drop occurs in the following areas:

- Distribution fluid to heat pipe evaporative section
- Drop from boiling to condensing sections of heat pipe
- Condensing fluid to finned surface

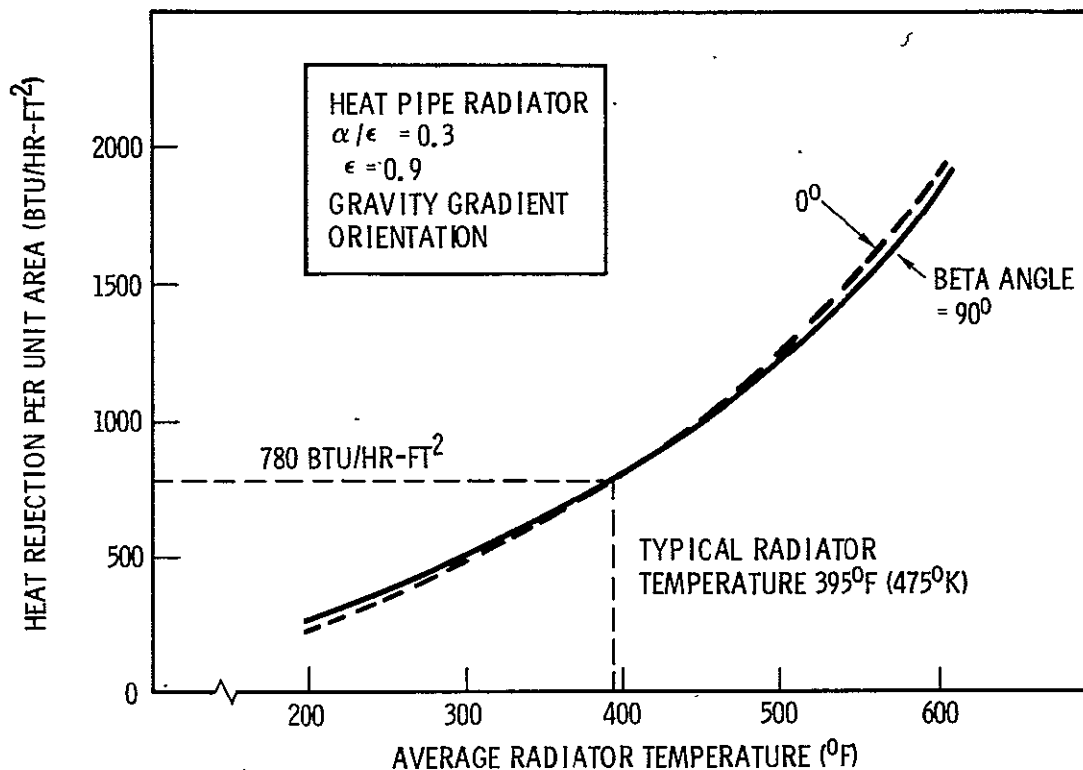


Figure 4-29. Reactor Brayton Radiator Performance

Based on this 475 K (395° F) average radiator temperature, about 2.46 kW/m<sup>2</sup> (780 Btu/hr/ft<sup>2</sup>) can be rejected. Based on a heat rejection requirement of 360 kW, a radiator area of 146 m<sup>2</sup> (1,575 ft<sup>2</sup>) is required.

From Figure 4-29, it can be seen that beta angle has a negligible effect on the radiator performance. This is a result of two considerations: (1) high heat influxes from albedo are picked up at low beta angles, and (2) direct solar heat influxes are large and occur over the entire orbit at high beta angles. These have opposite impact on performance; thus, the beta angle effects are small.

The effect of radiator coating characteristics was examined, and it was found that performance is little affected by this parameter. As  $\alpha_s/\epsilon_t$  varied from 0.18 to 0.36, performance varied up to 5.7 percent.

Radiator Design Analyses. The following sizing analyses were made to arrive at duct and heat pipe characteristics: (1) sizing the distribution duct, (2) sizing the heat pipes, and (3) determining the required interfaces between heat pipe and distribution fluid.

The distribution duct was sized based on two criteria: (1) sufficient velocity to prevent carbonization deposition of the Dowtherm A fluid; and (2) pumping power considerations.

A flow velocity of at least 0.61 m/sec (2 ft/sec) is required to meet the first criterion. Based on the heat transport and temperature requirements, a flow rate of 2,901 kg/hr (6,383 lb/hr) is required. An inside diameter of 4.37 cm (1.72 inches) results from these requirements. A relatively high Reynold's number results which is well in the turbulent region. The resulting pressure drop amounts to about 1.5 kg/cm<sup>2</sup> (21.3 psi) which includes cooler, distribution duct (length and bends), and heat pipe interface manifolds. A pump power requirement of 447 watts results. Based on these results, a tube design was selected with 4.37 cm (1.72 inches) inside diameter and a wall thickness of 0.51 mm (0.02 inches).

A simplified thermal analysis was performed on the interface between distribution fluid and heat pipe boiling surface. The purpose of this task was to design a feasible interface. The convective heat transfer coefficient between the Dowtherm A and tube wall is high, resulting in a temperature gradient of about 2.8° to 5.6°C (5° to 10°F), depending on the length of the interface (1 to 2 feet was used). This is a low value and is not a design driver.

Next, a simplified design was analyzed in which the distribution duct would be brazed, welded, or bonded directly to one side of the heat pipe. Results showed that a large temperature gradient would be needed to transfer the required heat between distribution fluid and the heat pipe wall. This temperature difference would be expected to be as large as 28° C (50° F) which would greatly decrease radiator performance. This results from the relatively thin distribution duct wall (0.51 mm or 0.020 inch) and low conductivity of stainless steel. Based on these results, a design was chosen in



which the distribution fluid flows directly over the heat pipe boiler section in a manifold or jacket-type design. This arrangement can reduce the temperature difference between distribution fluid and heat pipe to  $2.8^{\circ}$  to  $5.6^{\circ}$  C ( $5^{\circ}$  to  $10^{\circ}$  F).

Heat Pipes. The operating temperature range of the brayton cycle cooler (and thus the pumped intermediate cooling loop) dictated the use of mercury as the heat pipe working fluid. Mercury has a typical operating range of 450 to 900 K ( $810^{\circ}$  to  $1,620^{\circ}$ R) in heat pipe applications. Based on past industry experience, stainless steel was selected as the material for the heat pipe container and wick.

The number of heat pipes to be utilized was determined by the radiator geometry and reasonable fin effectiveness values; this resulted in 20.32-cm (8 inch) pipe spacing, for a total of 150 heat pipes. The resulting thermal load for each heat pipe was therefore 2.4 kW. To minimize the diameter of each pipe, a configuration was selected to use semicircular heat pipes with an evaporator section at the midpoint. This would result in a heat pipe whose effective length was one-half the total (semicircular) length and whose effective heat load was one-half the total heat load.

Utilizing a multiple-layer screen composite wick, the maximum capillary heat transport limits in zero gravity were determined as a function of heat pipe diameter: utilizing this information, the inside diameter of the heat pipes was determined to be 1.91 cm (0.75 inch) at the required heat transport rate of 4,147 W-m (163,266 W-inch).

System Weight. Characteristics discussed in the above paragraphs were used to determine the weight of the radiator and heat sink distribution system. The results were shown earlier in Table 4-6 and include redundant distribution loops and a radiator of  $200 \text{ m}^2$  ( $2,151 \text{ ft}^2$ ). Redundancy is included for vulnerable system elements; heat pipes are not made redundant since loss of a small percentage of these components will only slightly reduce total performance.

Heat Source Heat Exchanger. The heat source heat exchanger was scaled from the unit which was designed for the solar brayton unit. The method of scaling was based on the assumption that unit weight is proportional to the product of overall heat transfer coefficient and heat transfer area (UA). The UA requirements for the solar brayton unit were only slightly lower than for the reactor brayton even though about 52 percent less heat was transported. This occurred because the required effectiveness for the solar brayton was 0.87 compared to 0.56 for the reactor brayton. The resulting weight of the reactor brayton heat source heat exchanger was 652 kg (1,434 pounds). Pressure drop characteristics will be similar to the solar brayton unit.

#### Discussion of Results

A preliminary design has been generated and performance analytically verified for a heat pipe radiator and heat sink distribution system which interfaces with a 120-kWe reactor brayton power system. The design presented above will reject about 492 kW, whereas the design requirement is 360 kW. This additional power capacity, amounting to an extra 37 percent beyond requirements, is retained because: (1) some area may be required for an auxillary radiator, and (2) an allowance should be made for performance degradations if some heat pipes are damaged.

The calculated weights for the system are considerably higher than the numbers furnished by LASL. However, redundant equipment and the relatively high weight of mercury are the driving weight items.

#### 4.3.5 Reliability Analysis

The reliability and maintenance characteristics of a nuclear reactor brayton power system were determined for use in the SCB application. The reactor brayton system employs an advanced heat-pipe cooled nuclear reactor being considered for development by LASL for ERDA. The thermal energy generated will be used by a high-temperature brayton cycle engine which is a candidate converter for use with the high-temperature heat pipe reactor. The maximum electrical output of the reactor brayton power system is 120 kWe.

### Brayton Power Unit

The reliability and maintenance characteristics of the brayton power unit were analyzed first because the conversion units are common to both power systems being considered. The results are shown in Table 4-13.

Table 4-13  
BRAYTON POWER CONVERSION SYSTEM

| Item                                                                            | Basic<br>Failure Rate<br>( $10^6$ hr) | 10-Year Reliability    |                     |
|---------------------------------------------------------------------------------|---------------------------------------|------------------------|---------------------|
|                                                                                 |                                       | Without<br>Maintenance | With<br>Maintenance |
| Rotating machine                                                                | 2.0                                   | 0.8392                 | 0.8392              |
| Piping                                                                          | -                                     | 0.8878                 | 0.8878              |
| Controls                                                                        | 2.0                                   | 0.8392                 | 1.0000              |
| Gas management                                                                  | 1.6                                   | 0.8692                 | 1.0000              |
| Power conversion system                                                         |                                       | 0.5435                 | 0.7450              |
| 1 unit operating,<br>2 units standby redundancy,<br>0.99 switchover reliability |                                       | 0.9245                 | 0.9936              |

The analytical technique used was to determine a basic failure rate (or a mean time to failure) and apply this failure rate over the expected operating time of 10 years (87,600 hours). The basic failure rate of the rotating equipment was obtained by extrapolating gas turbine component experience for the compressor, turbine, alternator, and the other miscellaneous equipment. The results indicate a basic failure rate of  $2 \times 10^{-6}$  failures per hour. This results in a 10-year reliability of 0.8392. It was assumed that this portion of the brayton unit was not amenable to repair or maintenance; therefore, the reliability is the same for a maintenance or a non-maintenance assumption.

The reliability for the piping portion of the brayton unit was established by a count of the welds involved in fabrication and assignment of a weld failure rate of  $1 \times 10^{-9}$  failures per hour for an operating temperature about  $600^\circ$  F and a failure rate of  $1 \times 10^{-10}$  failures per hour for temperatures below  $600^\circ$  F.

The overall result of this analysis and calculation, as reported in Table 4-13, is a probability of no failure of 0.8878 with no possibility of maintenance. The failure prediction for the control system was based on analysis of similar systems. The gas management system reliability analysis also utilized previous work on brayton systems (Reference 4-4).

It was assumed for this analysis that the maintenance operations will be perfect; therefore, all maintainable items (controls, gas management) have a reliability of 1.0 in the "with maintenance" situation.

Table 4-13 shows that the reliability prediction with no maintenance is 0.5435 and with maintenance is 0.7450. It is planned to install three brayton units in the power system with two in a standby redundant configuration. Assuming a switchover reliability of 0.99, the reliability of the triple redundant system is 0.9245 without maintenance and 0.9936 with maintenance.

#### Reactor Brayton System

The results of the reliability analysis of the reactor brayton system are shown in Table 4-14. The thermal energy generated by the reactor is transmitted to the brayton unit by means of 91 heat pipes and a heat exchanger.

The loss of an individual heat pipe would not have a significant effect on power output because the heat generated in the fuel element/heat pipe module would be transmitted to the adjacent heat pipes by conduction. It was assumed, however, that a loss of 20 percent would cause sufficient local hot spots to require a substantial drop in power. Therefore, this was established as the failure level. A basic failure rate of 0.1 failures per million hours was established which gave a 10-year reliability of 0.9358. It was also assumed that the system could operate with a failure of one control drum (either full in or full out). The basic failure rate for the control drum is the failure rate for the stepping motor. The failure probability for the control system was obtained from similar units. It was assumed that the control system could be maintained, but not the control drums or heat pipes.

The failure probability for the intermediate heat exchanger was based on previous analysis (Reference 4-4) which was determined using the weld count technique described above. Two redundant heat exchangers were assumed.

Table 4-14  
REACTOR BRAYTON POWER SYSTEM

| Item                | Operating Units | Basic Failure Rate (10 <sup>6</sup> hr) (10-Yr Reliability) | 10-Yr System Reliability |            |
|---------------------|-----------------|-------------------------------------------------------------|--------------------------|------------|
|                     |                 |                                                             | No Maint                 | With Maint |
| Reactor,            | 1               |                                                             | 0.8416                   | 0.8508     |
| Heat Pipes          | 72/90           | 0.1                                                         | 0.9358                   | 0.9358     |
| Control Drums       | 5/6             | 1.0                                                         | 0.9092                   | 0.9092     |
| Control System      | 2/3             | 0.7                                                         | 0.9891                   | 1.0000     |
| Heat Exchanger      | 1/2             | 0.9080                                                      | 0.9915                   | 0.9915     |
| Power Conversion    | 1/3             | 0.5435                                                      | 0.9245                   | 0.9936     |
| Radiator System     | 1/2             | 0.9411                                                      | 0.9965                   | 0.9965     |
| Radiator            | 1/1             | 0.9484                                                      |                          |            |
| Pumps               | 1/2             | 1.0                                                         |                          |            |
| Instrumentation     | 100 sets        |                                                             | 0.9998                   | 0.9999     |
| Sensors             | 2/5             | 0.1                                                         | 0.9999                   | 0.9999     |
| Signal Conditioning | 2/5             | 0.1                                                         | 0.9999                   | 1.0000     |
| System              |                 |                                                             | 0.7686                   | 0.8352     |

The radiator portion of the dual output radiators was also analyzed with the weld count method. In addition, a probability of zero meteoroid puncture of 0.9867 was applied. Each of the two output radiator loops have two redundant fluid pumps with a basic failure rate of 1.0 failures per million hours. The results show that each radiator has a 10-year probability of no failure of 0.9411. The two radiators in parallel (each can carry full load) have a success probability of 0.9965.

It was assumed that 100 parameters will be monitored. A two-out-of-five success logic (i. e., allowing three failures) was assumed for each parameter measurement to keep the overall system immune from sensor failures. It was assumed that the signal conditioning could be maintained but the actual sensors could not.

The overall results show a 10-year reliability of 0.7686 for the no-maintenance case and of 0.8352 for the maintenance case.

### Results

Table 4-15 shows the results of the reactor power system analysis. If we assume a standby redundant configuration, the overall probability of at least one out of two reactor brayton power modules operating properly is 0.9645 without maintenance and 0.9838 with maintenance. In order to achieve a level of about 0.999, a total of four modules would be required if no maintenance is allowed, and three units if maintenance is assumed.

Table 4-15  
10-YEAR RELIABILITY

| Units* | Reactor Power Unit (120 kW) |                  |
|--------|-----------------------------|------------------|
|        | No Maintenance              | With Maintenance |
| 1/1    | 0.7686                      | 0.8352           |
| 1/2    | 0.9645                      | 0.9838           |
| 1/3    | 0.9952                      | 0.9985           |
| 1/4    | 0.9994                      |                  |

\*Assumes standby redundancy

### Maintenance Analysis

The maintenance analysis (Table 4-16) was conducted by determining the number of failures expected on a probabilistic basis for each of the maintainable items. In addition, the average time to accomplish the repair or replacement was estimated. As shown in Table 4-16, the time allocation is made up of four separate estimates: (1) a wait time defined as time spent putting on and taking off a space suit; (2) the time used in detecting which component failed; (3) the actual time spent in repairing or replacing the component; and (4) the time involved in checkout or adjustment of the repaired or replaced component. The maintenance hours over 10 years for each system is then the product of the number of failures and the maintenance hours per failure.

Table 4-16  
MAINTENANCE ANALYSIS

| Item                | Maintenance<br>Hr/Fail | Reactor System |          |
|---------------------|------------------------|----------------|----------|
|                     |                        | Fail/10 Yr     | Hr/10 Yr |
| Power Conversion    |                        |                |          |
| Controls            | 4.2*                   | 0.53           | 2.23     |
| Gas Management      | 4.2                    | 0.42           | 1.76     |
| Instrumentation     |                        |                |          |
| Signal Conditioning | 4.2                    | 4.38           | 18.4     |
| Reactor Power Plant |                        |                |          |
| Control System      | 4.2                    | 0.18           | 0.77     |
| Unit Totals         |                        | 5.51           | 23.16    |
| *Wait Time          | 2.5 hr                 |                |          |
| Detect              | 0.2                    |                |          |
| Repair/Replace      | 1.0                    |                |          |
| C/O and Adjust      | 0.5                    |                |          |
| Total               | 4.2 hr                 |                |          |

The results show that the reactor system will have about six failures in the 10-year period with a maintenance time of 23 hours.

#### Reliability Optimization

The reliability goal for the power units (0.999) discussed above was established to approach the reliability level achieved in a previous study (Reference 4) on a basis of equivalence to solar arrays. It would appear that the reliability goal should be established on a firmer basis such as economics or safety. One method that has been used on other programs is indicated in Figure 4-30. This method determines the optimum economic level of reliability by determining the overall life-cycle cost as a function of reliability. This is accomplished by determining the cost to increase the reliability level and adding the cost of failures as a function of the unreliability. As shown by the two curves of Figure 4-30, the initial cost — and probably operational cost — rises as the reliability increases, but the cost of failures drops as the reliability increases. The cost of reliability rises because of the increase in the initial cost of redundancy, high-reliability parts, and

- DETERMINE OPTIMUM ECONOMIC LEVEL OF RELIABILITY
- DETERMINE SYSTEM LIFE COST VERSUS RELIABILITY
- DETERMINE COST TO INCREASE RELIABILITY
- DETERMINE COST OF FAILURES
  - COST OF MAINTENANCE
  - COST OF DOWNTIME
  - COST OF REPLACEMENT UNITS
- DETERMINE MINIMUM-COST POINT
- OPTIMUM RELIABILITY LEVEL

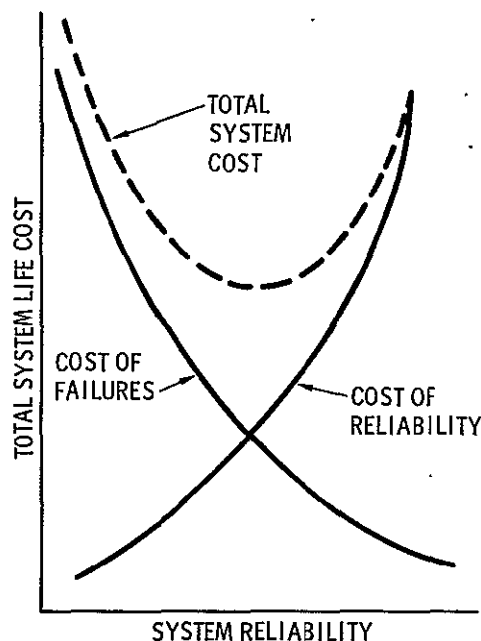


Figure 4-30. Reliability Optimization Analysis

other factors, and because of maintenance on additional components. The cost of failure drops with increasing reliability because fewer failures will occur. The cost of failure includes the cost of the downtime (including lost manhours and cost of replacement power), the cost of maintenance, and any other unique costs of failures. The minimum total of these two (the dashed line in Figure 4-30) occurs at the optimum reliability level.

#### 4.4 SUMMARY AND CONCLUSIONS

Our analysis of the influence of SCB integration requirements on a reactor brayton power system is summarized here.

Provisions for maintainability in power system design offer a reduction in the number of power modules required to attain a 10-year-mission life. Table 4-17 summarizes these findings, and emphasizes the appropriateness of a backup power module. Actual reliability of the proposed LASL design should be assessed when system design definition will permit in-depth analysis. A cost-effective reliability goal should be determined for the ERDA proposed power systems.



Table 4-17  
MAINTAINABILITY EFFECT ON RELIABILITY  
(REACTOR SYSTEM)

| Reliability Without Maintenance |        | Reliability With Maintenance |        |
|---------------------------------|--------|------------------------------|--------|
| <u>MOD</u>                      |        | <u>MOD</u>                   |        |
| 1                               | 0.7686 | 1                            | 0.8352 |
| 2                               | 0.9645 | 2                            | 0.9838 |
| 3                               | 0.9952 | 3                            | 0.9985 |
| 4                               | 0.9994 |                              |        |

Observations:

- Maintenance capability improves overall system reliability
- Shutdowns will occur, backup module is prudent
- Replacement time can range from 2 days to 30 days
- Actual reliability and redundancy is TBD - need complete design
- ERDA should determine power system cost-effective reliability for manned and unmanned missions.

The reactor brayton power system can meet the SCB's initial and growth power requirements based on the current system definition. The maximum power capability of a single power module is about 150 kWe unless the design can provide additional radiator area. These options for growth are shown below.

- A. Eliminate Orbiter docking module.
- B. Assess effects of increasing radiator temperatures.
- C. Operate two power modules at one time.
- D. Assess deployable radiator.

The reactor brayton power system is not limited to launch and assembly with an existing SCB. The design is acceptable as a free-flyer assembled in orbit by the Shuttle. Table 4-18 illustrates power system costs for a single reactor launch, the two reactor-single launch, and the free-flyer Shuttle assembly launch. The incremental cost of the free-flyer approach should be examined in light of a \$175 million cost for the 38-kWe power module which is not required in the free-flyer case.

Table 4-18  
REACTOR BASELINE AND OPTIONS

|                                      | Single Launch<br>(Baseline)                           | Dual Launch                                   | Free-Flyer<br>(Shuttle Erected)                                              |
|--------------------------------------|-------------------------------------------------------|-----------------------------------------------|------------------------------------------------------------------------------|
| First Launch                         | Reactor<br>Boom<br>Support Structure<br>Disposal Pack | Reactor (2)<br>Booms (2)<br>Support Structure | Reactor<br>Boom<br>Support Structure<br>Disposal Pack<br>Free-Flight Package |
| Second Launch                        | Same-less<br>Support Structure                        | Disposal (2)                                  | Same-Less<br>Support Structure &<br>Free-Flight Package                      |
| Costs<br>(in millions<br>of dollars) | Basic Reactor<br>Packages ~ \$189                     | Basic Reactor<br>Packages ~ \$189             | Basic Reactor<br>Packages ~ \$189                                            |
|                                      | Integration<br>Packages 14                            | Integration<br>Packages 14                    | Integration<br>Packages 14                                                   |
|                                      | Launches 40                                           | Launches<br>(1-1/4) 25                        | Free Flight Pkg<br>Launches 35<br>Shuttle Pwr Aug 40<br>2                    |
|                                      | Total: ~ \$243                                        | Total: ~ \$228                                | Total: ~ \$280                                                               |

Table 4-19 indicates supporting research and technology (SRT) requirements in light of the development risks for this system. In large measure, the table data reflect the conceptual design status at this time. A significant amount of analysis and testing will be needed before more conclusive assessments can be made.

Table 4-19  
REACTOR BRAYTON SYSTEM SRT REQUIREMENTS

- 
- REACTOR MATERIAL, 10-YEAR PERFORMANCE
    - MOLYBDENUM/SODIUM HEAT PIPES
    - FUEL/HEAT PIPE INTERFACE
    - HEAT PIPE INTERFACE WITH GAS LOOPS — SIZE, WEIGHT, COMPLEXITY, REDUNDANCY
    - EMERGENCY HEAT DUMP CAPABILITY — EFFECTS ON PACKAGING
  - BRAYTON ENGINE LAYOUT/PACKAGING FOR 120-kWe ENGINE
    - SINGLE OR REDUNDANT HEAT SOURCE HEAT EXCHANGE DESIGN
    - ANCILLARY EQUIPMENT (LIMITED VOLUME ON 2 REACTOR LAUNCH)
  - SHIELD
    - DESIGN TO MAINTAIN MIN/MAX TEMPERATURE LIMITS
    - RADIATION LEAKAGE THROUGH HEAT PIPES/EFFECTS ON PACKAGING AND LOCATION OF SOURCE HEAT EXCHANGERS
  - SINK HEAT EXCHANGER TO RADIATOR INTERFACE
    - HEAT PIPES WITH PUMPED LOOPS OR REDUNDANT PUMPED LOOPS?
    - WEIGHT, FABRICATION, RELIABILITY?
  - COST EFFECTIVE SYSTEM RELIABILITY
    - NEED DETAILED SYSTEM DEFINITION
  - DISPOSAL OPERATIONS
- RELATIVE RISK — HIGH
-

## Section 5

### SOLAR BRAYTON SIZING AND INTEGRATION

The solar brayton power system design definition was provided by Mr. J. A. Heller of NASA Lewis Research Laboratory (LeRC). The initial configuration of the power system is shown in Figure 5-1. The dynamic machinery, heat receiver and storage elements, engine room, electrical equipment, and waste heat radiator are located behind the focal point of the paraboloidal concentrator structure. The system was predicated on the laboratory's continuing brayton engine development, and early design, fabrication, and test of system elements including a deployable mirror, a receiver/heat storage subsystem, and a low-temperature radiator. All elements of the current design are scaled-up versions of previous LeRC efforts.

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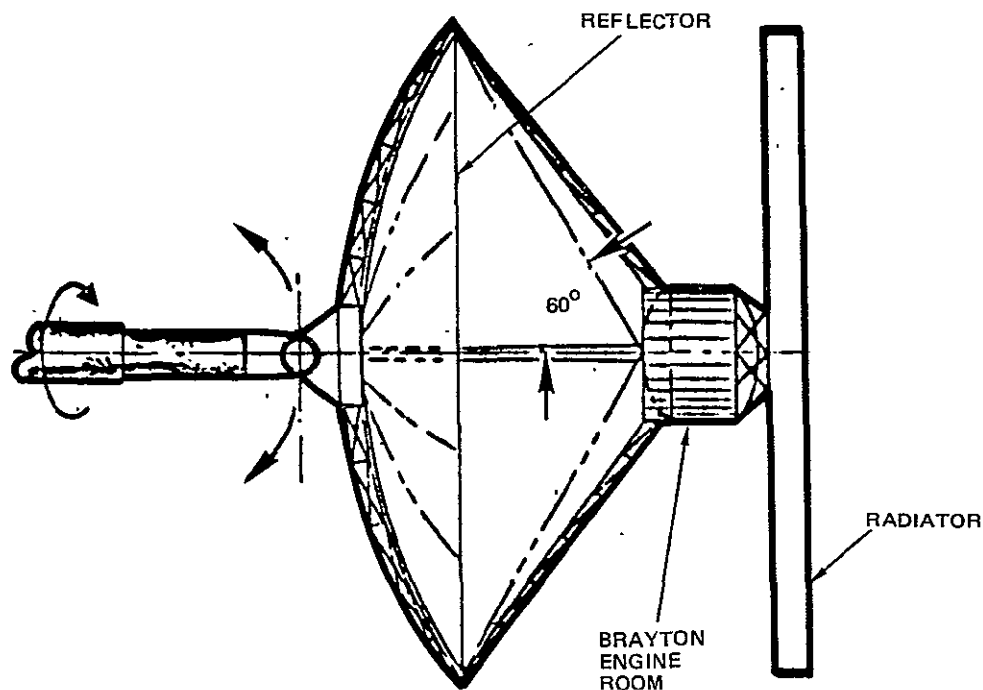


Figure 5-1. Initial Solar Brayton Configuration

## 5.1 DESIGN APPROACH

The design approach for the solar brayton was selected to provide engine redundancy sufficient for long life confidence, to simplify astronaut access so that maintenance could be accomplished at the replaceable black box level, and to eliminate the need for a peak power subsystem element. The decision to provide engine redundancy and to select maintenance at the black box level was based on previous Phase B Modular Space Station power system studies. The validity of the design approach was reaffirmed by reanalysis of the power system reliability, as discussed later in this section. The peak power element was eliminated simply to avoid the incremental cost of supplying and integrating other subsystem elements such as batteries, fuel cells, or energy wheels.

In addition, analysis was made to determine the number of power modules to be used and the number of engines to be operated per power module. The consequences of operating either one or two engines to meet the Space Construction Base (SCB) power requirement are summarized as follows:

- A. Operation of two engines to produce power
  - 1. Significant heat dump problem if one engine fails
  - 2. Increased maintenance
  - 3. Increased redundancy needed
  - 4. More difficult packaging problem
  - 5. Very complex heat source heat exchanger
  - 6. Lower engine efficiency.
- B. Operation of one engine to produce power
  - 1. Maximum engine efficiency
  - 2. Lower maintenance
  - 3. Less complex heat source heat exchanger.

One engine per module was selected to operate at average plus peak power, 60 kWe.

Brayton engine efficiency as a function of power rating is projected in Figure 5-2 from Reference 5-1, and the conceptual design and installation of the heat source heat exchanger and engine modules are shown in Figure 5-3. It is evident in the latter figure that either additional engines or additional heat

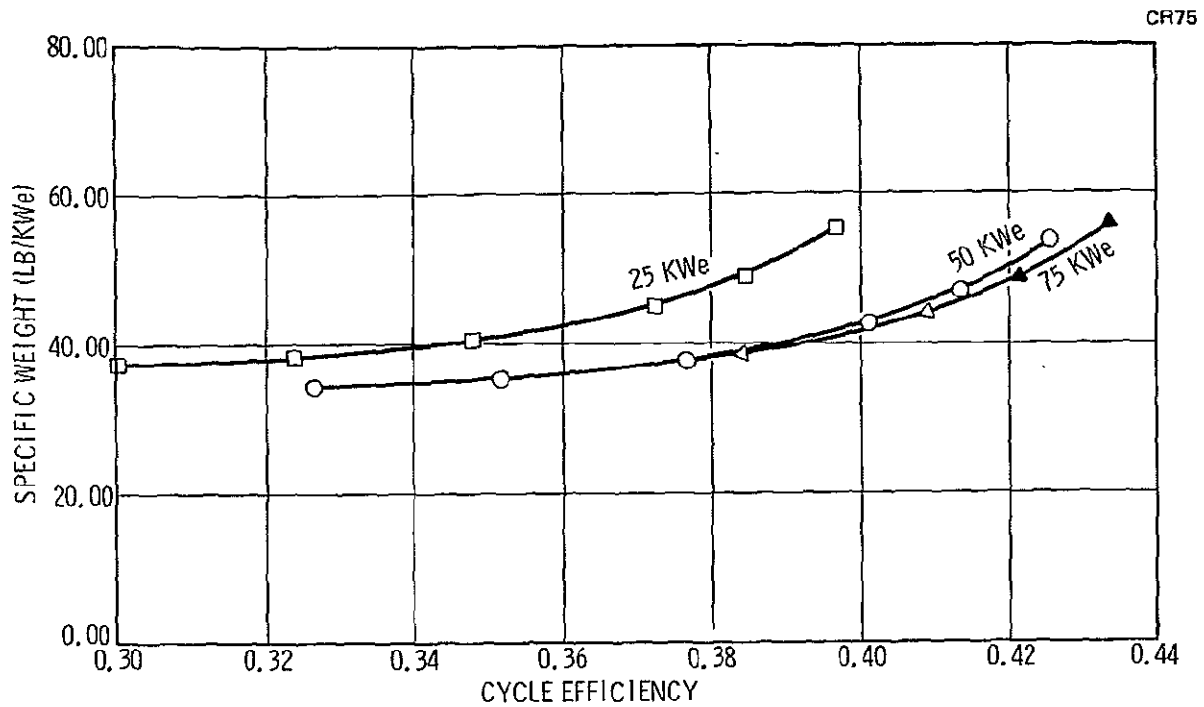


Figure 5-2. Brayton Engine Efficiency and Power Rating

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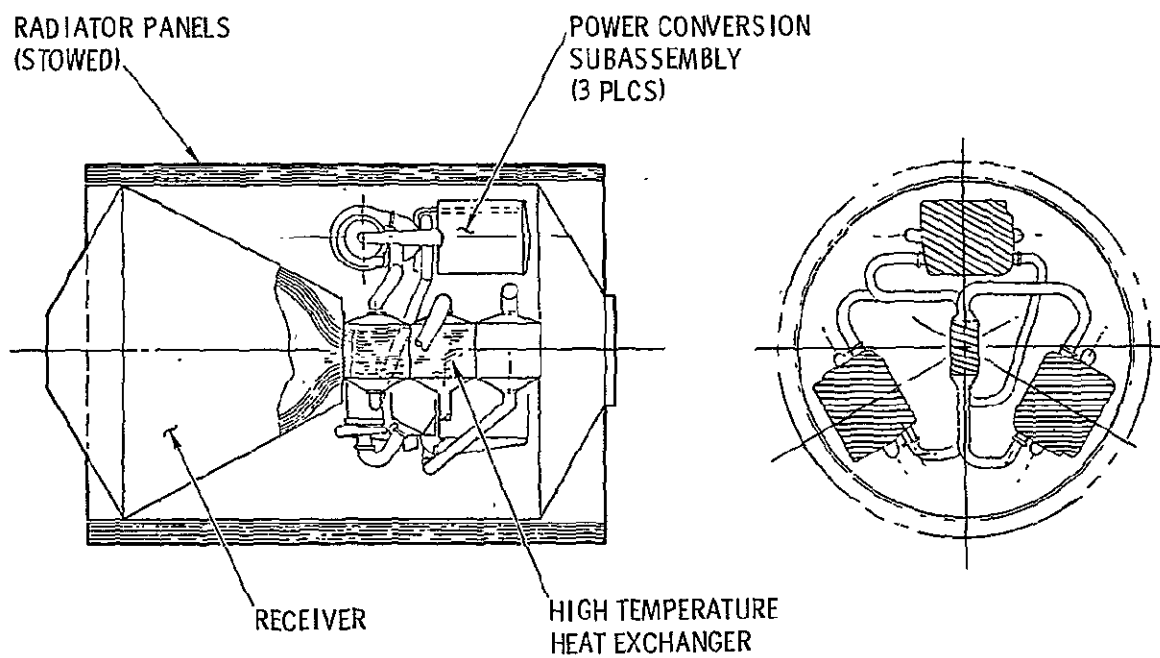


Figure 5-3. Solar Brayton Power Module - Engine Room

source heat exchangers will create installation problems. Operation of more than one engine to produce the nominal 60-kWe output would require additional engine installation to maintain high reliability, increase maintenance time, and cause a significant heat dump problem if failure leaves only one operable engine (i.e., 50% of the solar input would have to be dumped to space). Furthermore, the complexities of operating dual engines at either 50 or 100% of power were avoided to simplify the gas management system.

Inherent in operating one engine to produce the design power level is the requirement for rapid start-up of a standby engine. Since LeRC tests of the brayton B engine have repeatedly demonstrated start-up and power generation of these engines in less than 60 seconds, power interruption due to failure of a single engine will be minimal; therefore, single engine operation is recommended.

## 5.2 SYSTEM DESCRIPTION

### 5.2.1 Configuration Evolution

The initial solar brayton power module configuration (Figure 5-1) was examined to determine what modifications could be made to provide manned access for repairs, to simplify assembly of the square array radiator configuration, and to reduce the total mass supported at the focal point. Options included a cassegrainian configuration (Figure 5-4) and a receiver/heat-pipe configuration (Figure 5-5). The latter option retains only the receiver and heat storage elements at the focal point, and source heat is transferred approximately 30 feet from the receiver/heat storage elements to the engine room via heat pipes. Efficiency of heat transfer for this distance has not been determined; however, a fall-back position to a pumped loop mode is provided. The final selection, made with the concurrence of LeRC, is shown in Figure 5-6.

The selected configuration shows minor changes from the initial LeRC configuration. The primary differences are in the attachment point for the power module and in the radiator configuration which has been modified to avoid an on-orbit assembly operation. Concentrator shadowing from the attachment boom is less than 3% requiring the concentrator's intercepted solar flux to be

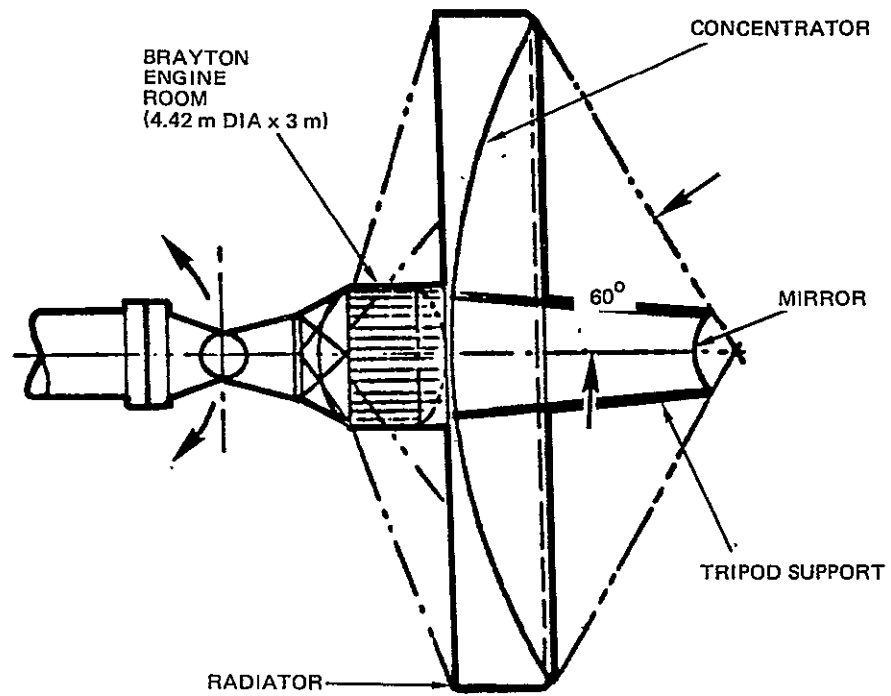


Figure 5-4. Cassegrainian Option

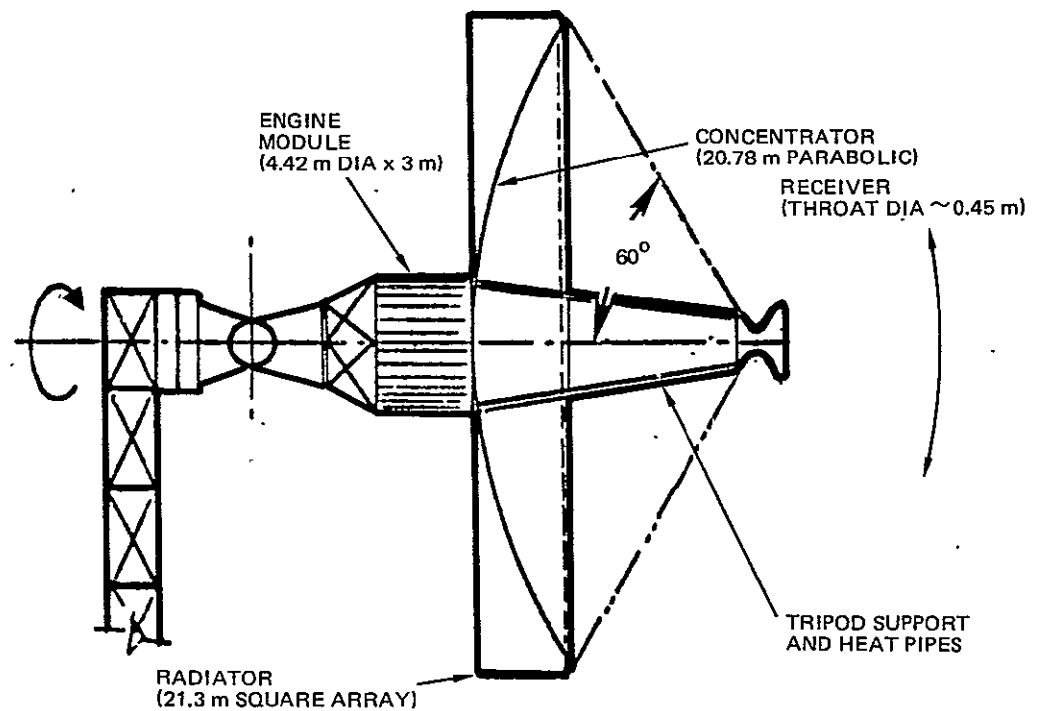


Figure 5-5. Receiver/Heat Pipe Option



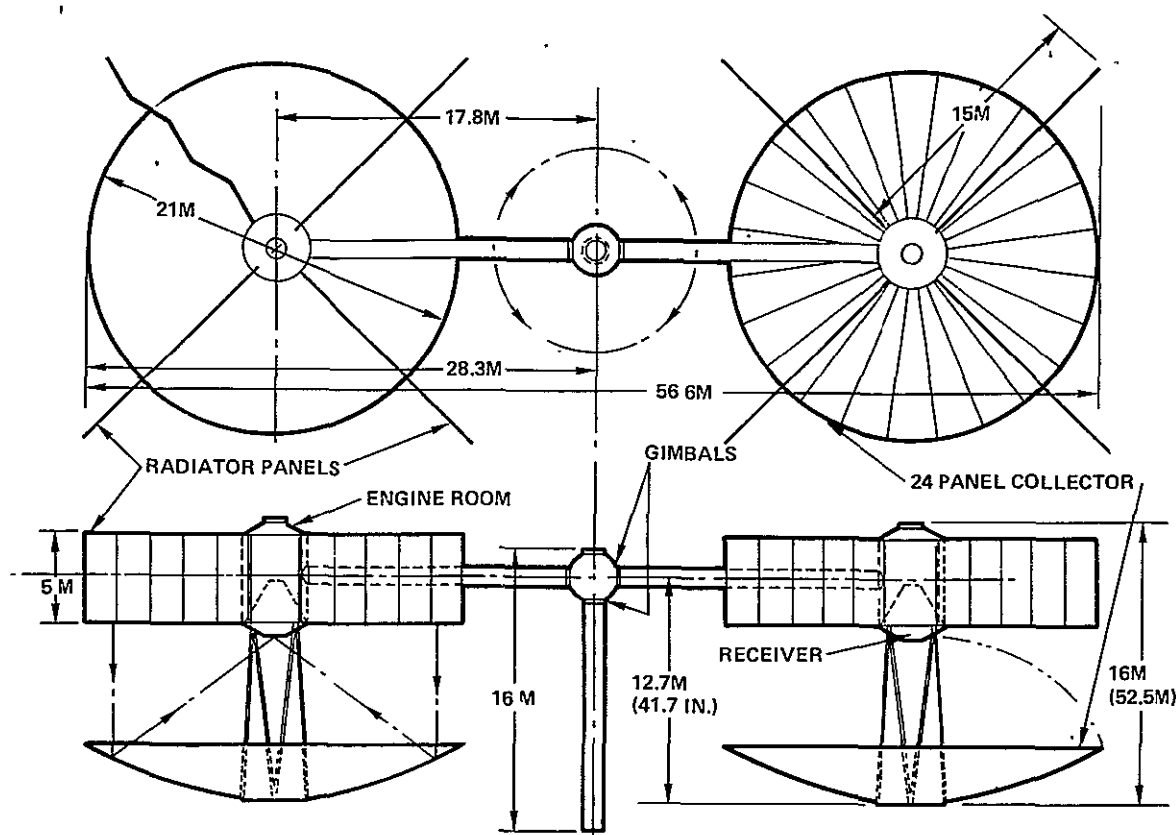


Figure 5-6. Solar Brayton Power System (120 kW)

slightly oversized. In relation to the other configurations, the selected configuration has the following advantages:

- Minimum mass supported on the structural extension
- Easy maintenance access
- Simplest deployment and assembly
- Simplest packaging into Orbiter.

The SCB configuration shown in Figure 5-6 would be flown with the x-axis in the local vertical, the y-axis along the velocity vector, and the z-axis perpendicular to the orbit plane. The interconnecting boom between the two power modules will accommodate the orbital viewing angle variation, and the main deployment mast will accommodate the cyclic beta angle variation with a maximum gimbal angle of  $\pm 95$  degrees. Actual gimbal angle requirements will be a function of the beta angle variation which is dependent on earth tilt (23.5 degrees) and orbit inclination.

Telescoping masts and booms are used in conjunction with a telescoping strongback structure to provide access by the construction crane for installation of the initial or replacement modules.

### 5.2.2 System Performance

The solar brayton paraboloidal concentrator intercepts solar flux and reflects that energy into the receiver/heat storage component which is located at the forward end of the engine module. The focused thermal flux impinges on the heat storage and heat transfer components which transfer approximately 51% of the maximum energy input to the electrical generator on a continuous basis. During the eclipse period, heat is removed from the lithium fluoride as that material undergoes phase change to the solid state. The solar brayton schematic diagram shows the cycle temperature variation and the helium-xenon gas flow path (Figure 5-7). With a receiver input of 438 kWt, the cycle

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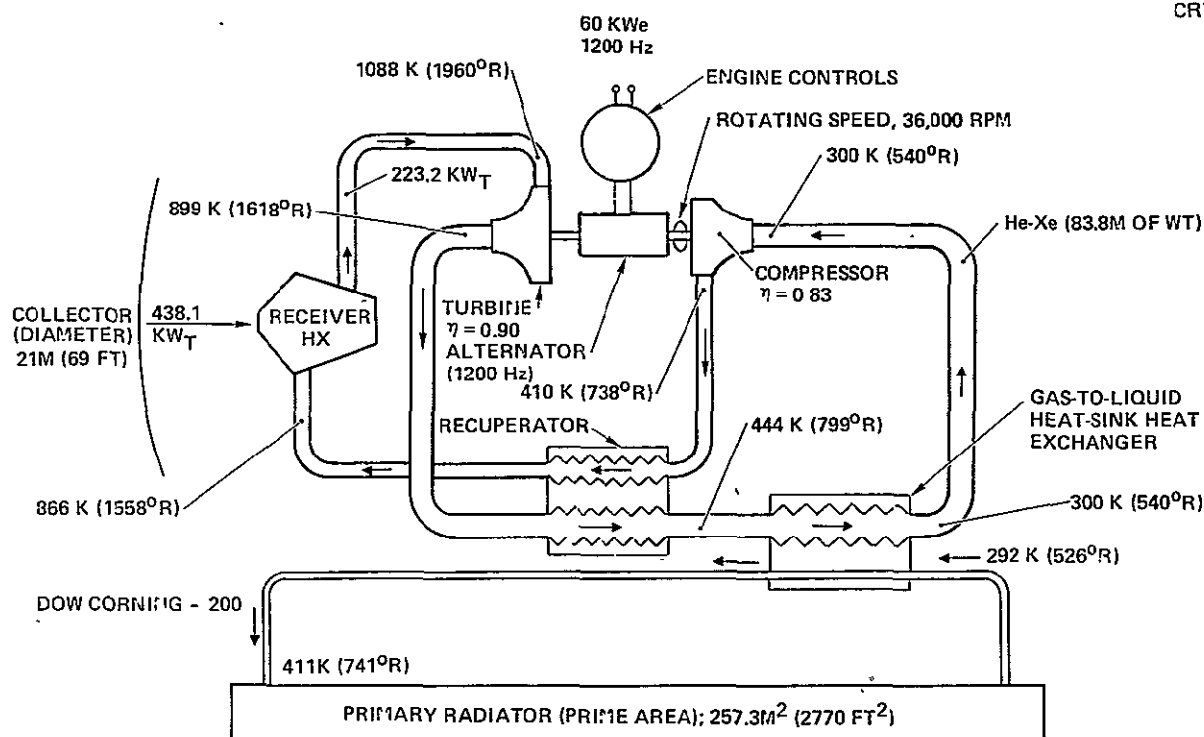


Figure 5-7. Solar Brayton Schematic

requires approximately 223 kWt to produce the 60 kWe power rating. The maximum cycle temperature of 1088 K (1500°F) is well within the state of the art, and the cycle efficiency is 26.89%. Further engine optimization is projected in Reference 5-1, while Figure 5-8 presents these projections. At the current cycle temperature of 1088 K (1500°F) turbine inlet and 300 K (80°F) compressor inlet, the projections indicate potential cycle efficiencies of about 40%. This increase in efficiency could permit significant reductions in system weight and concentrator size for fixed power requirements. Conversely, a

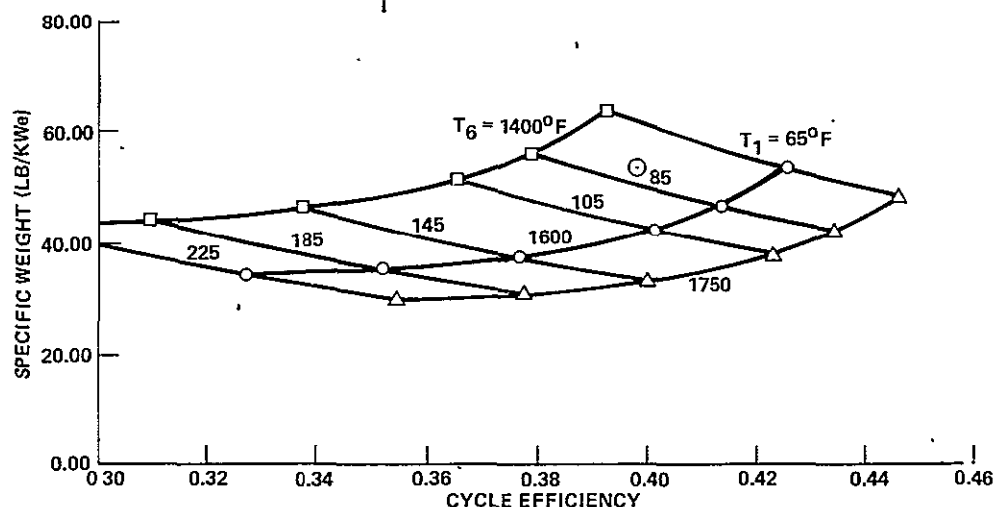


Figure 5-8. Brayton Technology Projections

significant power level growth would be possible if the system's physical dimensions are retained. Subsequent study should assess these potential performance improvements and the resultant cost impact.

Physical characteristics of the solar brayton power module are shown in Table 5-1. Component and launch weights for the two power modules, including integration equipment and contingency weight allowances, are given in Table 5-2 for the baseline mission.

Table 5-1

### SOLAR BRAYTON CHARACTERISTICS

|                                  |                    |                     |
|----------------------------------|--------------------|---------------------|
| Collector                        |                    | 730 kg (1,610 lb)   |
| Diameter                         | 21m                |                     |
| Surface Area                     | 375 m <sup>2</sup> |                     |
| Focal point (from apex)          | 9.1 m              |                     |
| Depth (apex to rim)              | 3m                 |                     |
| Radiator                         |                    | 1,225 kg (2,700 lb) |
| Area                             | 257 m <sup>2</sup> |                     |
| Panel (4)                        | 13m x 5m           |                     |
| Engine Weight (1)                |                    | 952 kg (2,100 lb)   |
| Brayton Rotating Unit            | 136 kg             |                     |
| Recuperator/Waste Heat Exchanger | 816 kg             |                     |
| Receiver                         |                    | 2,249 kg (4,960 lb) |
| Tubes with Lithium Fluoride      | 2,023 kg           |                     |
| Structure                        | 226 kg             |                     |

Table 5-2  
SOLAR BRAYTON WEIGHT SUMMARY

|                                       | Kilograms | Pounds |
|---------------------------------------|-----------|--------|
| Collector                             | 730       | 1,610  |
| Radiator                              | 1,225     | 2,700  |
| Concentrator Support                  | 57        | 125    |
| Brayton Engines (3 sets)              | 2,857     | 6,300  |
| Receiver/Lithium Fluoride             | 2,250     | 4,960  |
| Parasitic Load/Radiator               | 82        | 180    |
| Electrical Controls                   | 50        | 110    |
| Heat Source Heat Exch (3)             | 567       | 1,250  |
| Module                                | 218       | 480    |
| Subtotal                              | 8,034     | 17,715 |
| Double Gimbal } 1st Launch            | 770       | 1,700  |
| Boom }                                | 430       | 950    |
| Mast }                                | 370       | 820    |
| Mast + Adapter - 2nd Launch           | 430       | 950    |
| Contingency                           |           |        |
| First Module                          | 2,405     | 5,300  |
| Second Module                         | 2,120     | 4,670  |
| Orbiter Docking Module                | 1,770     | 3,900  |
| <hr/>                                 |           |        |
| First Launch = 13,780 kg (30,385 lb)  |           |        |
| Second Launch = 12,350 kg (27,235 lb) |           |        |

### 5.2.3 Integration Equipment

Integration equipment for the solar brayton power module consists of the deployment mast, booms, and gimbal drives (Figure 5-9). The main telescoping mast incorporates a backup and simplified gimbal at the boom and strongback interface to eliminate the risk of a single point failure. Both the main extension mast and the two extension booms have telescoping capability to permit initial placement and replacement. The gimbal drive mechanisms are repairable as designed. Orbit rate is accommodated in the two cross boom gimbals, which are driven independently. Beta angle accommodation is accomplished with the main mast gimbal, which is along the x-axis.

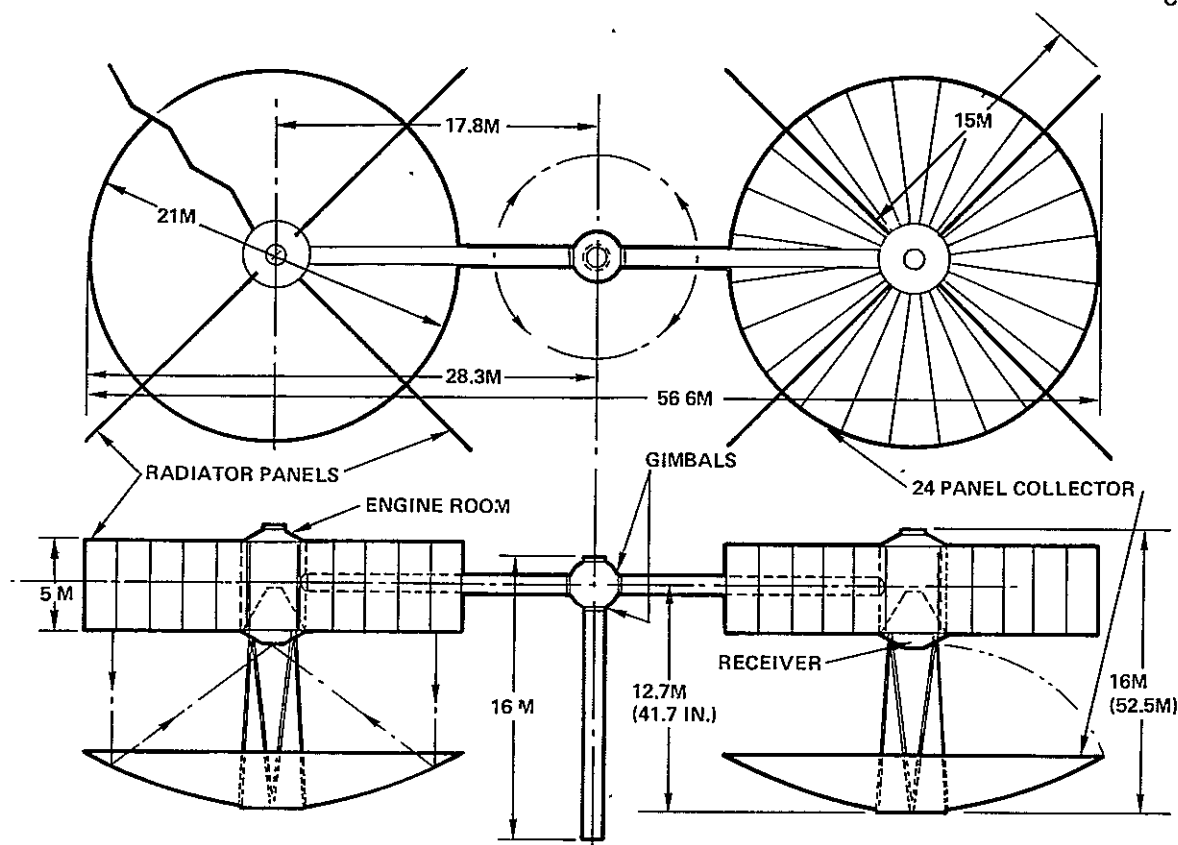


Figure 5-9. Solar Brayton Power System (120 kWe)

#### System Drive and Orientation Mechanism

The turret for the power system orientation is mounted on the end of a telescoping tunnel attached to the SCB strongback (Figure 5-10). The shell is machined from thick-walled, forged hemispheres of 2219-T87 aluminum which are welded together. Three identical harmonic drive assemblies are bolted to O-ring-sealed machined flanges located on the truncated shell. The main support mast is bolted to one of these assemblies, and a telescoping boom section is attached to each of the other two assemblies. The total assembly is composed of seven separate details: adapter section, outer truncated cone, load balancing cone, inner cone, flex spline, circular spline, and wave generator. The flex spline, circular spline, and wave generator comprise the harmonic drive, which has been selected for its high torsional stiffness, low backlash and tooth pad, and high single-step reduction.

#### Telescoping Booms

The telescoping boom sections are composed of 2219-T87 aluminum and have a 1.02-meter (40 inch) minimum inside diameter and an maximum extended length of 14 meters (45.93 feet), with the main mast extension being slightly

in length. (The assembly procedure for deploying and attaching the power system to the SCB is discussed in Section 5.3.1.)

#### Engine Room and Receiver

The engine room with installed engines will provide the structural base for the receiver, radiator, and tripod support for the collector mirror. This module is a maximum of 4.06 meters (13.3 feet) in diameter and 6.0 meters (19.6 feet) in length. It accommodates three power conversion subassemblies with high temperature heat exchangers arranged as shown in Figure 5-12. The receiver end of the structure incorporates the receiver/heat storage element and the structural base for the tripod support for the collector.

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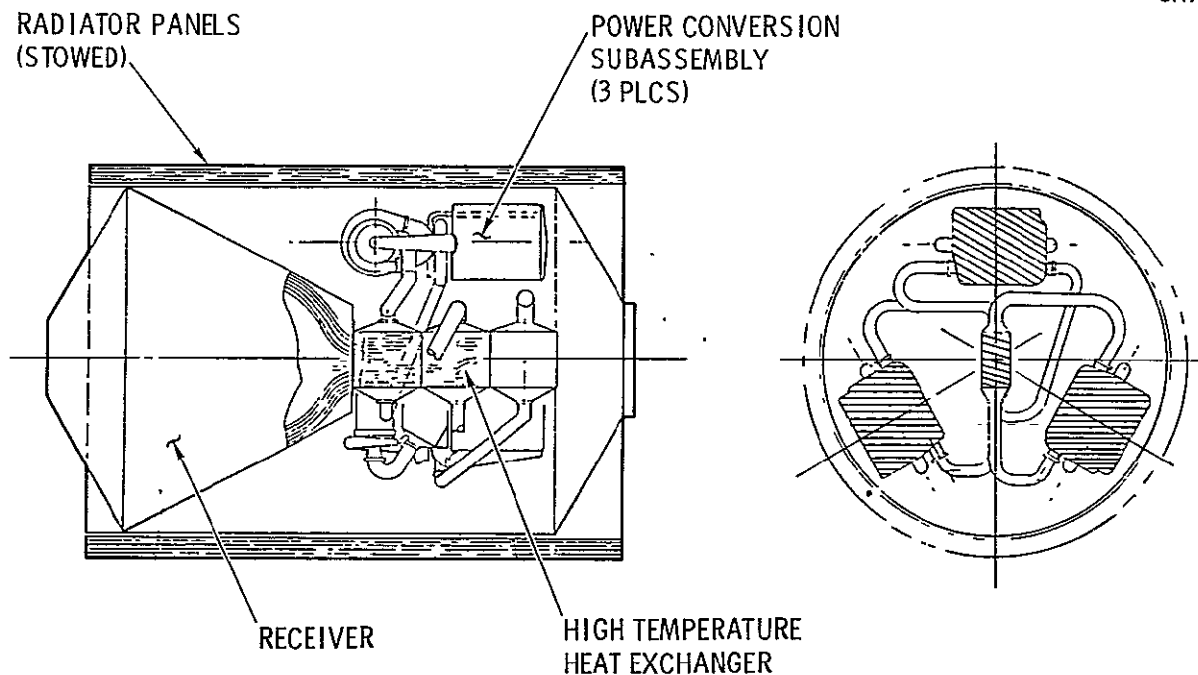


Figure 5-12. Solar Brayton Power Module - Engine Room

#### 5.2.5 Solar Brayton Components

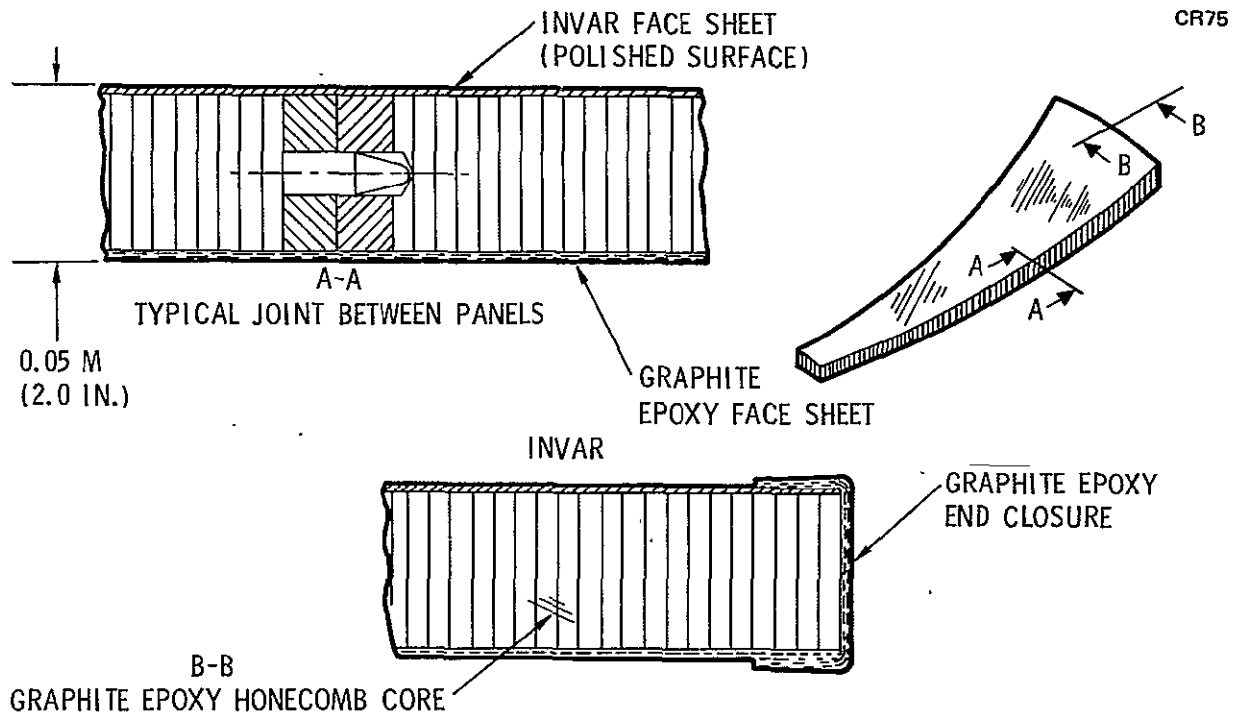
The solar brayton power module consists of the following elements, which are discussed sequentially.

- Collector/mirror
- Receiver/heat storage unit
- Brayton cycle engine and heat exchangers
- Radiator
- Support equipment

The integration equipment, booms, mast, and gimbals are discussed in Section 5.2.3.

### Collector/Mirror

The collector/mirror is a 21.03-meter (69-foot) diameter parabolic configuration, consisting of 24 petal segments each 9.4 meters (30.8 feet) in length. Design of the collector has not been finalized at this time and the weights given in Table 5-2 are based on use of a lightweight metal mirror. However, the light/dark thermal cycle of each orbit makes it appropriate to consider an alternate design which is less sensitive to thermal cycling. Such an alternate is shown in Figure 5-13. Each segment is constructed of



**Figure 5-13. Collector/Mirror Detail**

graphite epoxy honeycomb with a stretch-formed, polished invar face sheet for the reflective surface and a graphite epoxy outer sheet. The graphite epoxy material was selected for its extremely small coefficient of expansion and its stiffness. The former characteristic will help minimize thermal expansion and contraction problems.

Four petal segments are attached to the apex support ring by a hinge mechanism, and are hinged open with assistance from the SCB crane and EVA crew-

men. The remaining petal segments are placed in proper position and attached first to the support ring and then to each adjacent segment.

Preliminary weight estimates and credible weight variations for the honeycomb structure, which is an alternate to the metal concentrator described in Table 5-2, are indicated in Table 5-3.

The most significant problem appears to be with the structural assembly of the paraboloidal segments to assure surface matching and to avoid distortions in the assembly. With the current system design, it is apparent that launch weight will not be limiting; thus, a development program can consider a wide range of attachment and alignment concepts.

#### Receiver/Heat Storage Unit

The configuration for the receiver/heat storage unit shown earlier in Figure 5-3 shows the interface between the heat pipe heat exchanger concept and the brayton engines. The concept was developed from earlier design and fabrication of an 11 kWe receiver for LeRC (Figure 5-14), and was based on circulating the engine working fluid (helium-xenon) directly through the heat receiver tubes. Ducting into and out of the tubes was accomplished using ring-shaped manifolds. A cross-section view of the early receiver design, details of the heat storage material (lithium fluoride) encapsulation on each tube, and location of the receiver aperture and the heat dump control doors are shown in Figures 5-15 and 5-16.

The 10-year mission lifetime for the SCB dictates a capability for more than one engine assembly in order to make credible long-term reliability determinations. This need for more than one engine makes the gas receiver tubes and manifolds less desirable since valving would be needed to isolate operating and standby engines. Consequently, LeRC has tentatively selected heat pipes to transfer the solar energy directly to the brayton cycle engines, as shown in Figure 5-3. The heat storage encapsulation concept shown in Figure 5-16 would be retained, with heat pipes replacing the individual gas tube assemblies shown in Figure 5-15. By changing to a heat pipe configuration, it is possible to consider a variable conductance heat pipe configuration which could replace the heat dump doors. A charge of non



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Table 5-3  
COLLECTOR/MIRROR WEIGHTS

| Description               | Lower Limits               |                         | Nominal                    |                         | Upper Limits               |                         |
|---------------------------|----------------------------|-------------------------|----------------------------|-------------------------|----------------------------|-------------------------|
|                           | Thickness<br>or<br>Density | Mass (kg)               | Thickness<br>or<br>Density | Mass (kg)               | Thickness<br>or<br>Density | Mass (kg)               |
| INVAR Face Sheet          | 0.51 mm                    | 1,539                   | 0.70 mm                    | 2,154                   | 0.70 mm                    | 2,154                   |
| Core                      | 24 kg/m <sup>3</sup>       | 458                     | 32 kg/m <sup>3</sup>       | 610                     | 40 kg/m <sup>3</sup>       | 762                     |
| Graphite Epoxy Face Sheet | 0.61 mm                    | 0.399                   | 0.61 mm                    | 399                     | 0.91 mm                    | 598                     |
| End Closure/miscellaneous |                            | 56                      |                            | 56                      |                            | 56                      |
|                           |                            | 2,452 kg<br>(5,406 lbm) |                            | 3,219 kg<br>(7,096 lbm) |                            | 3,570 kg<br>(7,870 lbm) |

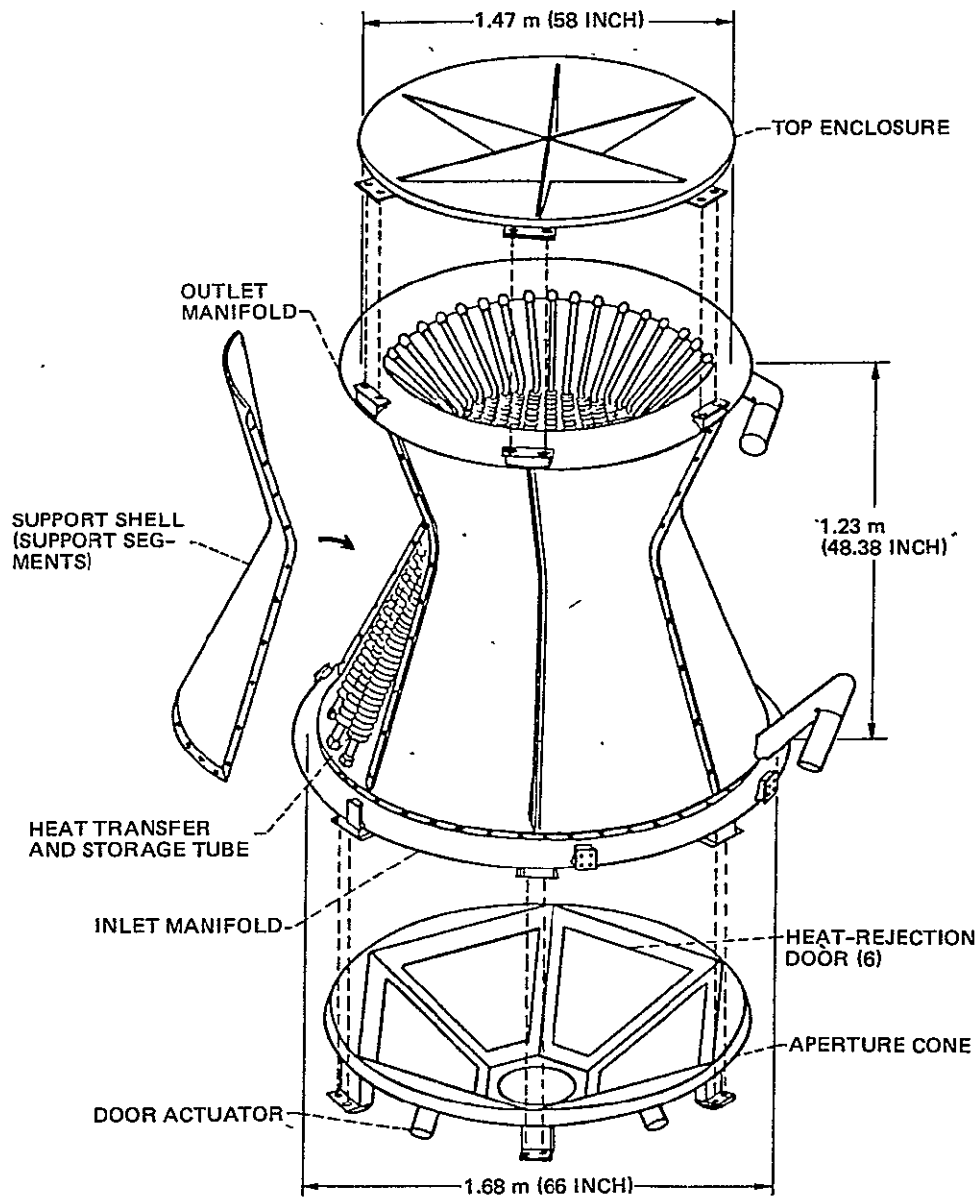


Figure 5-14. Brayton Heat Receiver Design

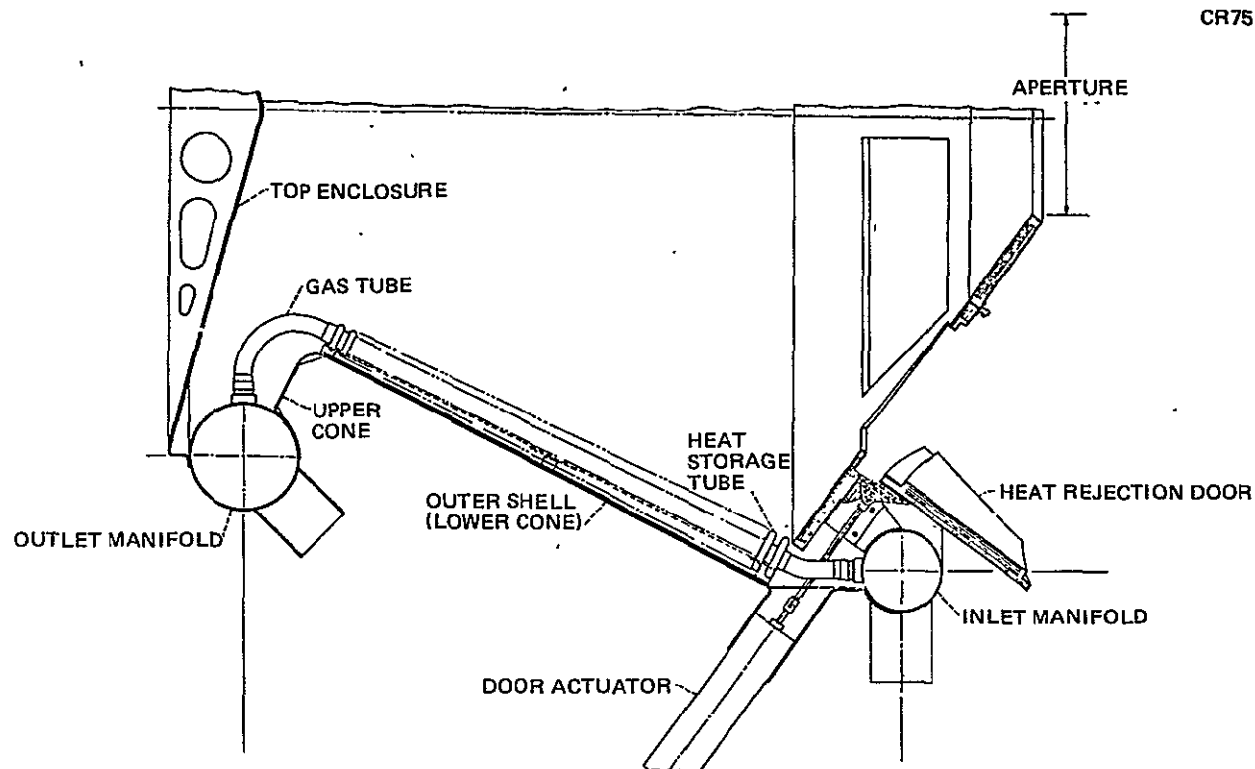


Figure 5-15. Solar Receiver Cross Section

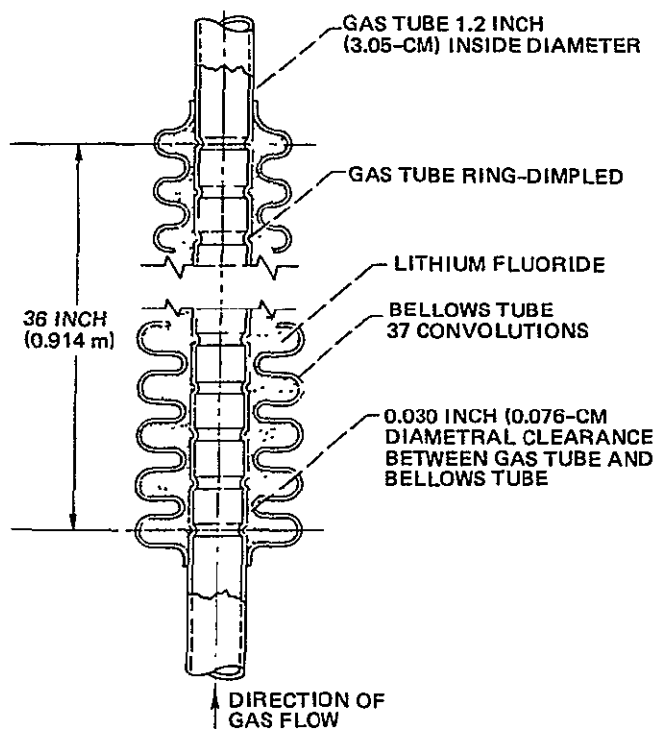


Figure 5-16. Heat Transfer and Storage Tube

noncondensing gas is added to one end of each heat pipe. Then, whenever excess solar input occurs (e.g., at high beta angles and high orbit inclinations), the working fluid vapor pressure increases, displaces the normal vapor interface, and uncovers additional heat pipe condenser area. This concept is shown in Figure 5-17 and is reported in Reference 5-2.

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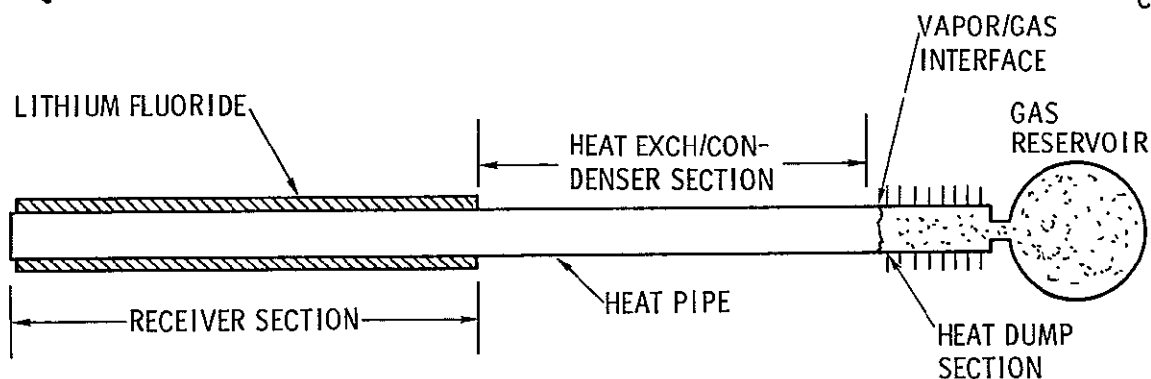


Figure 5-17. Variable Conductance - Heat Control

According to Reference 5-2, the variable conductance heat pipe configuration can reduce fluctuations in the system operating temperatures and since the control technique is passive, there are advantages to the concept in terms of reliability. The same conceptual design will probably be incorporated into the heat pipe reactor design.

#### Brayton Cycle Engine and Heat Exchangers

The brayton cycle engine includes the brayton rotating unit (BRU), the heat source heat exchanger, the waste heat recuperator, and the heat sink heat exchanger (Figure 5-7).

A smaller version of the brayton engine, developed by LeRC and called the B engine, has been in operation at a NASA LeRC facility and has accumulated 30,000 hours of operation as of 1 July 1977. Operation has been largely unattended and has accomplished much in gaining recognition of the capabilities of the closed-cycle gas turbine power system.

The BRU consists of a radial inflow single-stage turbine, a Lundell-type alternator, and a radial outflow single-stage compressor on a single shaft. The shaft is supported on two journal and one double-acting thrust bearings which are self-acting gas bearings that utilize the working fluid. No development problems are known that would impede design and fabrication of a scaled-up rotating unit for the SCB power system.

For this study MDAC has developed a preliminary brayton heat source heat exchanger design based on a 50-heat pipe configuration. The concept includes three discrete heat exchangers each approximately 0.76 meters in length and approximately 0.6 meters by 0.3 meters in cross-sectional areas (Figure 5-3) and incorporates 50 heat exchanger pipes, each with an outside diameters of 3.18 cm. The pipes are arranged in 11 rows with 5 pipes in 6 rows and 4 pipes in 5 rows.

The waste heat recuperator is a single-pass counterflow gas-to-gas heat exchanger with stacked plate-fin surfaces. It has been designed for low gas flow pressure drops and high heat exchanger effectiveness. This unit is also extremely adaptable in that it can be optimized for various power levels without major redesign by increasing or decreasing the number of plate fins in the stack.

The heat sink heat exchanger employs a cross-counterflow arrangement with plate-fin surfaces in both the gas and liquid flow paths. The liquid coolant, an organic silicone fluid, makes eight passes across the straight-through gas flow.

The waste heat recuperator and the heat sink heat exchanger were designed as a single package (BHXU — brayton heat exchanger unit) for the system, but can also be designed as separate units with little or no weight penalty.

#### Radiator

The four radiator arrays (each 13 by 5 meters) are segmented and folded in pleated fashion against the engine room module to meet the maximum payload diameter available in the Orbiter bay. The tube/fin structure has been sized with sufficient thickness to withstand stowing without permanent deformation. The cruciform array was selected to simplify both launch packaging and deployment and will require flexible connections at the hinge lines of each segment. As designed, the radiator area is conservatively sized with an excess area of about 38 percent. Interconnections between folded segments will use flexible pigtails at each hinge line.

The radiator will use redundant radiator loops with two circulating pumps in each loop; either loop can dissipate the cycle waste heat. The circulating

coolant is DC-200, which was selected in part to avoid freezing during system shutdown.

### Support Equipment

The power module's electrical subsystem includes a speed control, a voltage regulator, a parasitic load, dc power supply, inverters, and a signal conditioner. The BRU has been designed for constant speed steady-state operation at 36,000 rpm. To maintain this constant output frequency of 1,200 Hz, a speed control and a parasitic resistive load are used. The parasitic load absorbs any unused power. The voltage regulator maintains the three-phase voltage at 120 volts, line-to-neutral, or 208 volts, line-to-line. During steady-state operation, the dc power supply rectifies part of the three-phase alternator output to provide the internal needs of the power system at +28 volts. The engine controls, signal conditioner, and inverters use dc power. The inverters supply 400 Hz power to the SCB distribution system and to the liquid pump motors used in the heat rejection/radiator system.

### 5.3 INTEGRATION

The power system has been assessed in terms of impact on SCB operations, guidance and control, and thermal environment. In addition, a special analysis was conducted to assess solar brayton reliability and to establish confidence that this power system has lifetime prospects equivalent to the other systems. This reliability/maintainability analysis is discussed in this section.

The main integration effort was directed at the Shuttle-tended SCB configuration as shown in Figure 4-14 in Section 4. A 38 kWe power module provides electrical services while either the baseline solar array power platform or the solar brayton system is assembled. After in-orbit assembly of the SCB power system, the power module is released to function as needed in other Shuttle-supported missions.

During buildup of SCB power systems, the hardware elements depicted in Figure 4-14 provide electrical power, the services of the construction crane, and the construction capability of the strongback fixture. For this mission definition a single power module can be assembled, installed, and operated

with minimum risk until a second power module can be launched, assembled, and installed.

An alternate mission mode has also been examined to determine the incremental costs and requirements if the solar brayton is launched as an initial free-flyer with assembly done by the Orbiter crew, using the remote manipulator system (RMS). This mission option assumes operability of the power system before launch of any SCB elements and eliminates the need for the 38 kWe power module.

These two mission modes were assessed by the same people who developed the operational timelines for fabrication/assembly/deployment of the SCB configurations, mission hardware elements, and the SCB baseline power platform. Consequently, the comparative analyses are consistent in approach. The results are documented and unique requirements and impacts of the free-flyer power module are identified in the following section.

#### 5.3.1 Operations Analysis

Construction of the solar brayton power system was analyzed, first, by considering utilization of the SCB capability and, second, by considering Shuttle sortie capability. It was found that both approaches are feasible, but use of the sortie mode requires additional capabilities and one more launch than would be required by the SCB.

#### Baseline Buildup

The baseline buildup sequence using the SCB is summarized in Figure 5-18. In this buildup the first power module package is delivered and installed on a turntable on one end of the SCB strongback. The tunnel/gimbal assembly stowed in the center of the package (see Section 5.2 for a description of the packaging) is removed using the SCB crane and installed on the other end of the strongback. The four outer "leaves" of the mirror reflector packages are then deployed and locked in place. The remaining mirror segments are then installed, one at a time. Each segment is attached at one point to the collector support ring and at the two outer corners to the two adjacent mirror segments. The single tie at the collector ring has two degrees of freedom, allowing the angle of the segment to be varied with respect to the





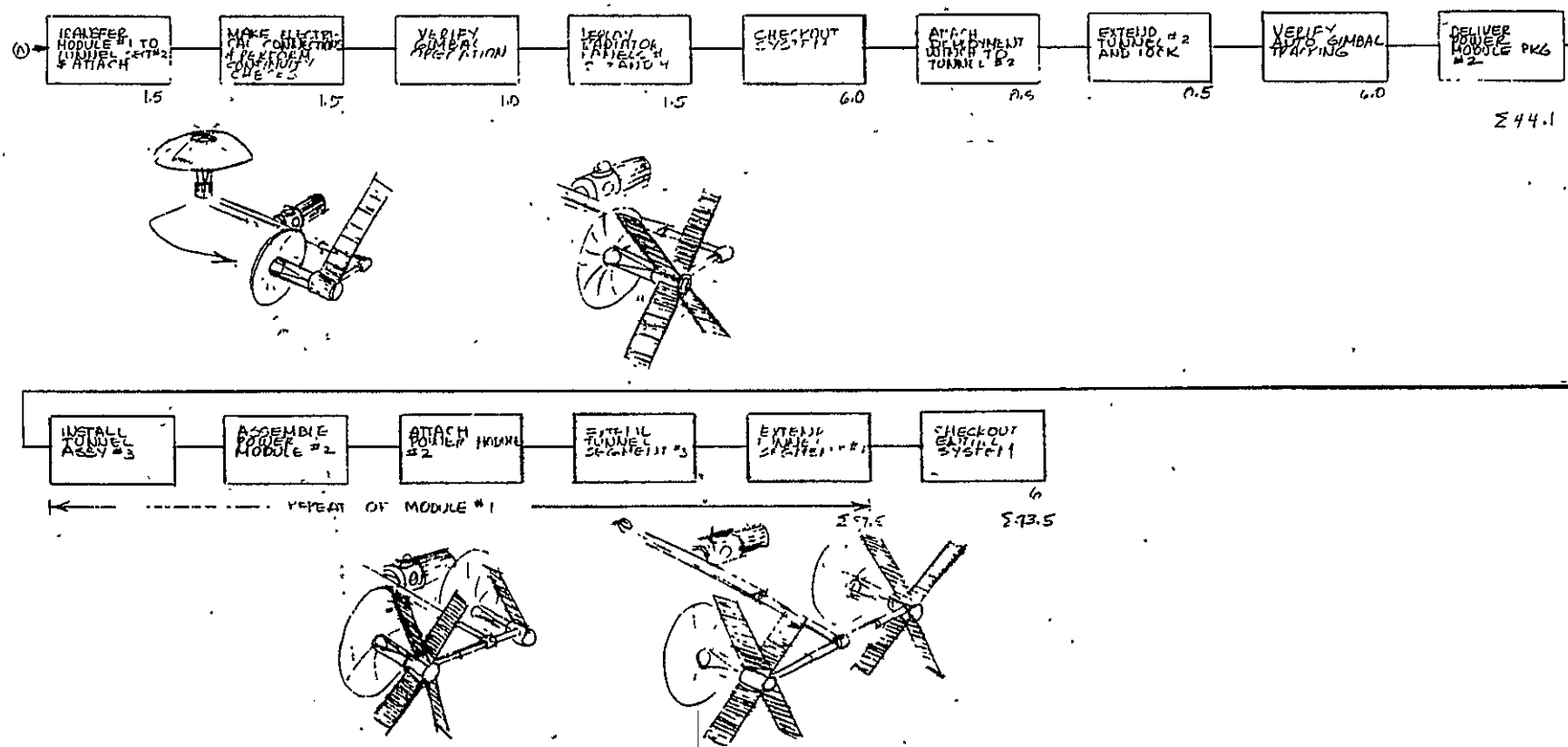


Figure 5-18. 120-kWe Solar Brayton Power Module System Build-Up (Two 60-kWe Launch Packages) (Sheet 2 of 2)

ring. The outer ties are adjustable to allow alignment of the mirror segments.

Once the reflector is assembled, alignment aids are installed, the protective coatings on the mirror segments peeled off, and the mirror segments adjusted to focus columnated light on the receiver. This requirement for alignment was pursued in great depth; however, no simple method was uncovered, and additional work is clearly warranted in this area.

At this point in the assembly, one of the radiator panels is deployed to uncover the fitting, which allows the module to be attached to the tunnel. The module is then transferred to the tunnel, installed, the remaining radiator panels deployed, and the system checked out. After checkout the tunnel is extended to allow full rotation of the module. The second power module is then delivered and assembled in a similar manner.

The time required to set up an individual power module is relatively short – four days assuming two work shifts per day. Assuming the second module is delivered in a timely manner, the whole operation takes about eight days, as illustrated in Figure 5-19. The majority of the work requires EVA, with the

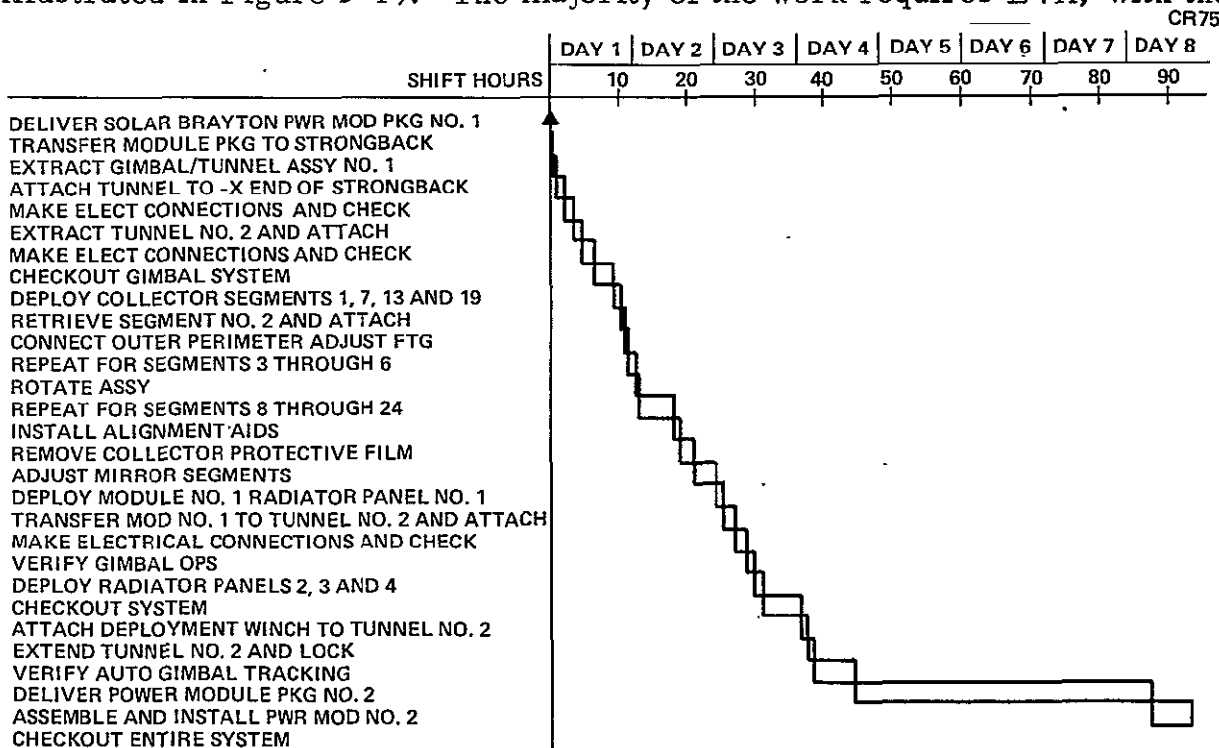


Figure 5-19. 120-kWe Solar Brayton Power Module Build-Up Timeline

most critical EVA tasks being assembly of the reflector and making and checking out mechanical/electrical interfaces. The most sensitive task is transfer of the assembled module, with one radiator panel deployed, from the strongback to an operational position. This is done using the crane with extreme care to avoid damaging the system.

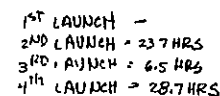
#### Sortie Mode Buildup

An optional buildup approach in which the system is assembled in a Shuttle sortie mode is depicted in Figure 5-20. Packaging and the assembly sequence of the module is similar to assembly accomplished by using the SCB, though some operational differences are noted.

To accommodate subsequent buildup of the SCB, the capability to dock to the power module and, while docked, berth a space construction module to it is necessary. It was determined that the only way to accomplish this was by use of a docking adapter delivered with the power module. Since this increases the length of the delivered assembly, a Shuttle docking module cannot be included in the bay; thus, the first Shuttle flight merely delivers the module and leaves it on orbit where a module mounted control system provides stability for subsequent docking.

The second Shuttle docks to the end port of the docking adapter and the power system is assembled. Because there is no cherry picker platform system on the Shuttle, some simple scaffolding, anchored to the cargo bay, must be erected to allow EVA astronauts access to the outer attach points on the mirror. Also, because the RMS does not have adequate reach, some form of manipulator must be included to help position the reflector segments. Access to the collector support ring requires that EVA mobility aids be incorporated in the module and support truss structure. Assembly of the mirror segments requires the reflector assembly to be rotated. This capability will have to be included in the power module, possibly as part of the collector support ring.

Subsequent to module assembly the Orbiter separates and the third Orbiter comes up and docks to a side port on the adapter. The SCM is assembled on the payload installation and deployment aid (PIDA) and berthed to the end-port. Since the interfacing port between the SCM and the adapter must accommo-



# OBIGENTIA PREGHIERA OF POOR QUALITY

**Figure 5-20. Solar Brayton Power System Sortie Mode Delivery**

date both docking and berthing, this dual mode of operation capability is required.

A fourth launch then brings up the strongback and gimbal assembly and the module is installed in its operating position. The Shuttle returns the docking adapter to earth. The second power module is assembled using the SCB as discussed earlier.

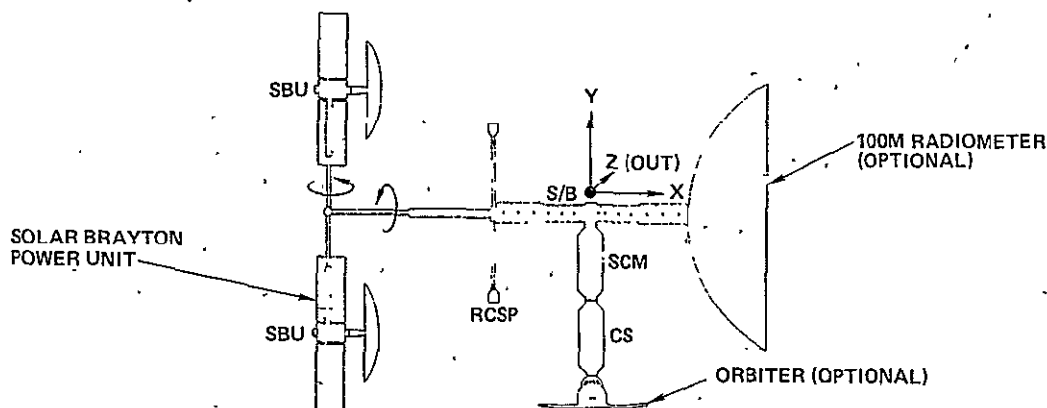
#### Comparison of the Baseline with the Sortie Mode Buildup

Assembly of the solar brayton power module is operationally similar in both construction approaches. However, of the two approaches, the utilization of the Shuttle in a sortie mode to construct the first power module suffers by comparison since the sortie mode imposes the following additional requirements:

- One additional Shuttle flight
- Use of a docking adapter
- Development of a dual mode, berthing/docking port
- Addition of a stability and control system to the power module
- Inclusion of a capability to rotate the reflector relative to the base of the power module
- Incorporation of EVA mobility aids in the power module
- Inclusion of erectable scaffolding in the cargo bay
- Development of a scaffold-mounted manipulator to assist in assembly.

#### 5.3.2 Guidance and Control Analysis

The guidance and control analysis for the solar brayton system has considered two solar brayton units at one end of the strongback, as shown in Figure 5-21. Both units operate simultaneously and must be pointed toward the sun to an accuracy of under 15 arc minutes. The solar brayton units (SBUs) are gimballed first about the x-axis and then about the mast orthogonal to it. Additional vernier control of each unit will probably be required. The study considered the presence or absence of the 100-meter radiometer at the other end of the strongback, representative of one of the largest mission hardware elements considered at this time in the Space Station study. This allowed the consideration of the effect of the extremes in size, mass, and



• CONFIGURATIONS STUDIED (WITHOUT & WITH ORBITER)

- BASIC
- BASIC WITH SOLAR BRAYTON UNITS (SBU'S) ROTATED 90° OUT OF PLANE
- BASIC + 100-M RAD
- BASIC + 100-M RAD WITH SBU'S ROTATED 90°
- CONCEPTUALIZED AN SBU VERNIER POINTING SYSTEM

Figure 5-21. Solar Brayton Attitude Control Analyses

aerodynamic force on stability, control, and orbit-keeping of the vehicle. Additionally, the study considered the presence or absence of the Orbiter. Although the Orbiter will nominally be attached as a logistics vehicle only for small percentages of the time, its presence was considered for sake of completeness and to give additional insight into the effect of its large mass and displacement of the center of gravity from the docking axis.

Orientation requirements of the orbiting vehicle depend upon accommodation of the program mission hardware and the desirability to reduce the cost and resources necessary to maintain the vehicle in orbit. It has been established that any mission hardware (or SBU) requiring highly accurate pointing will contain the instrumentation and actuators to satisfy that requirement. The SCB would then provide the coarse pointing of the element, and stabilize the entire vehicle against the environmental torques. If the nature of the mission hardware is such that no particular orientation is required, then an orientation requiring minimum resources is permitted.

A stable orientation exists relative to the vertical and orbit plane as the result of gravity gradient torques and gyroscopic coupling with orbit rate. This is illustrated in Figure 5-22. This orientation will generally be the

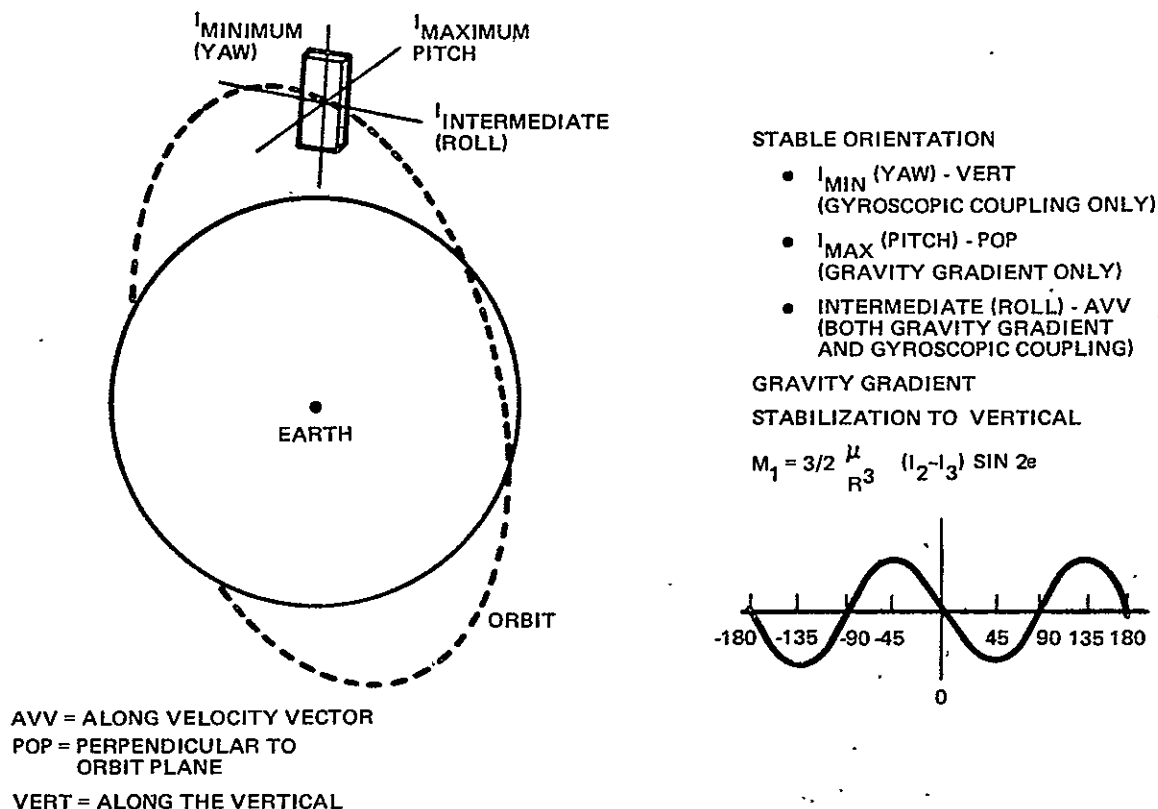


Figure 5-22. Gravity Gradient Stabilization with Gyroscopic Coupling

preferred long-term, low-resource orientation unless the associated drag necessitates an excessively large amount of propellant for orbit-keeping. The characteristics of the principal axes for the stable gravity gradient orientation are as follows:

- Roll. The roll principal axis is directed along the velocity vector. The value of the moment of inertia about this axis,  $I_R$ , is intermediate to the values about the two other principal axes.
- Pitch. The pitch principal axis is directed perpendicular to the orbit plane. The value of the moment of inertia about this axis,  $I_p$ , is the largest of the three principal axes.
- Yaw. The yaw principal axis is directed along the vertical. The value of the moment of inertia about this axis,  $I_Y$ , is the smallest of the three principal axes.

Orientation other than this will require the expenditure of resources (such as attitude control propellant) to hold orientation against the gravity gradient torques. These alternate orientations will come from particular mission hardware test, calibration, or data gathering operations. Because of

peculiarities of a given configuration, it may not be possible to assume the stable gravity gradient orientation. The next best solution would be a zero moment, but unstable gradient, orientation obtained by a 90 degree rotation about any of the principal axes from the stable orientation. This requires a limited amount of active control about the unstable axes.

### Analysis

The analysis of the moments of inertia of the solar brayton configuration of Figure 5-21 is summarized in Tables 5-4 through 5-12. The first column of each table is an indication of the rotation of the principal axes from the geometrical axes. The numbers given are the magnitude of the rotation from the indicated axis, but do not include the full 3 by 3 matrix of inverse direction cosines. The I and  $\Delta I$  columns are the moment-of-inertia parameters of the configuration upon which the rest of the columns are based. Because yaw and roll in orbit coordinates are coupled through gyroscopic torques, the next three columns are presented as if each of the three axes is an uncoupled pitch axis. They represent the maximum torque per axis (at 45 degree orientation of the axis from the vertical), the torque per degree from the null point, and the "capture" velocity (the velocity below which the vehicle will oscillate about the stable point). Low capture velocities require a tight attitude control before release of the vehicle to stable free oscillation. The last three columns represent the period of oscillation for small perturbations, assuming that each of the axes is stable as a pitch, yaw, or roll axis, respectively. The numbers boxed represent the applicable values assuming complete, three-axis gravity gradient stabilization.

The gravity gradient data for the basic configuration of Figure 5-22 with and without the 100-meter radiometer and with and without the Orbiter are provided in Tables 5-4 through 5-7. The solar brayton units will have to be rotated about the vehicle x-axis as a function of position of the vehicle in a given orbit. Data for the units rotated 90 degrees are given in Tables 5-5 and 5-7. It is apparent that a single orientation will not suffice for a stable gravity gradient condition, although the  $x_p$ -axis is nearly always vertical. A low beta condition (beta is defined as the angle between the orbit plane and the sun line) requires the solar brayton units (SBUs) to be located normal to the orbit plane at "sunrise/sunset" and in-plane at high noon. A sequence



Table 5-4  
CONFIGURATION: SOLAR BRAYTON IN-PLANE

| Axis                   | Rotation<br>(deg) | $I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | $\Delta I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | $T_{max}^*$<br>(ft-lb) | $T/\theta^*$<br>(ft-lb/deg) | $\dot{\theta}^*$<br>(deg/sec) | $P_p$<br>(min) | $P_y$<br>(min) | $P_r$<br>(min) |
|------------------------|-------------------|-----------------------------------------------|------------------------------------------------------|------------------------|-----------------------------|-------------------------------|----------------|----------------|----------------|
| <u>Without Orbiter</u> |                   |                                               |                                                      |                        |                             |                               |                |                |                |
| $X_p$                  | 13.7              | 8.571                                         | 8.12                                                 | 15.27                  | 0.53                        | 0.109                         | 55.5           | 96.1           | 45.1           |
| $Y_p$                  | 13.5              | 21.023                                        | 20.57                                                | 38.69                  | 1.35                        | 0.110                         | 54.5           | 94.5           | 47.3           |
| $Z_p$                  | 6.3               | 29.138                                        | 12.45                                                | 23.43                  | 0.82                        | 0.073                         | 82.6           | 143.0          | 71.5           |

Conclusions: GG stable orientation is:  $X_p$  - VERT,  $Y_p$  - AVV,  $Z_p$  - POP

With Orbiter

|       |      |        |       |       |      |       |      |       |      |
|-------|------|--------|-------|-------|------|-------|------|-------|------|
| $X_p$ | 32.2 | 45.944 | 42.69 | 80.32 | 2.80 | 0.107 | 56.0 | 97.0  | 48.5 |
| $Y_p$ | 36.4 | 23.655 | 20.40 | 38.39 | 1.34 | 0.103 | 58.1 | 105.7 | 50.3 |
| $Z_p$ | 21.7 | 66.348 | 22.29 | 41.93 | 1.46 | 0.064 | 93.1 | 161.3 | 80.7 |

Conclusions: GG stable orientation is:  $X_p$  - AVV,  $Y_p$  - VERT,  $Z_p$  - POP

\*Given as in-plane (Pitch) characteristics

Note: Orbit altitude = 450 km

AVV = along velocity vector

POP = Perpendicular to orbit plane

VERT = Along the vertical

Table 5-5  
CONFIGURATION: SOLAR BRAYTON POWER UNITS ROTATED 90°

| Axis                   | Rotation<br>(deg) | $I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | $\Delta I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | $T_{max}^*$<br>(ft-lb) | $T/\theta^*$<br>(ft-lb/deg) | $\dot{\theta}^*$<br>(deg/sec) | $P_p$<br>(min)                                     | $P_y$<br>(min)                                      | $P_r$<br>(min)                                     |
|------------------------|-------------------|-----------------------------------------------|------------------------------------------------------|------------------------|-----------------------------|-------------------------------|----------------------------------------------------|-----------------------------------------------------|----------------------------------------------------|
| <u>Without Orbiter</u> |                   |                                               |                                                      |                        |                             |                               |                                                    |                                                     |                                                    |
| $X_p$                  | 11.0              | 8.708                                         | 1.42                                                 | 2.68                   | 0.094                       | 0.045                         | 133                                                | <span style="border: 1px solid black;">231.2</span> | 115.6                                              |
| $Y_p$                  | 28.8              | 25.724                                        | 15.59                                                | 29.33                  | 1.02                        | 0.087                         | <span style="border: 1px solid black;">69.3</span> | 120.1                                               | 60.1                                               |
| $Z_p$                  | 26.9              | 24.300                                        | 17.016                                               | 32.01                  | 1.12                        | 0.093                         | 64.5                                               | 111.7                                               | <span style="border: 1px solid black;">55.9</span> |

Conclusions: GG stable orientation is:  $X_p$  - VERT,  $Y_p$  - POP,  $Z_p$  - AVV .

|                     |      |        |       |       |      |       |                                                    |                                                     |                                                    |
|---------------------|------|--------|-------|-------|------|-------|----------------------------------------------------|-----------------------------------------------------|----------------------------------------------------|
| <u>With Orbiter</u> |      |        |       |       |      |       |                                                    |                                                     |                                                    |
| $X_p$               | 36.7 | 46.535 | 37.39 | 70.33 | 2.46 | 0.100 | 60.2                                               | 104.3                                               | <span style="border: 1px solid black;">52.2</span> |
| $Y_p$               | 41.5 | 26.017 | 16.87 | 31.73 | 1.11 | 0.089 | 67.0                                               | <span style="border: 1px solid black;">116.1</span> | 58.1                                               |
| $Z_p$               | 29.1 | 63.403 | 20.52 | 38.60 | 1.35 | 0.063 | <span style="border: 1px solid black;">94.9</span> | 164.4                                               | 82.2                                               |

Conclusions: GG stable orientation is:  $X_p$  - AVV,  $Y_p$  - VERT,  $Z_p$  - POP

\*Given as in-plane (pitch) characteristics

Note: Orbit altitude = 450 km

AVV = along velocity vector

POP = Perpendicular to orbit plane

VERT = Along the vertical

Table 5-6

CONFIGURATION: SOLAR BRAYTON +100 MRAD. IN-PLANE

| Axis                   | Rotation<br>(deg) | $I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | $\Delta I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | $T_{max}^*$<br>(ft-lb) | $T/\theta^*$<br>(ft-lb/deg) | $\dot{\theta}^*$<br>(deg/sec) | $P_p$<br>(min) | $P_y$<br>(min) | $P_r$<br>(min) |
|------------------------|-------------------|-----------------------------------------------|------------------------------------------------------|------------------------|-----------------------------|-------------------------------|----------------|----------------|----------------|
| <u>Without Orbiter</u> |                   |                                               |                                                      |                        |                             |                               |                |                |                |
| $X_p$                  | 19.2              | 36.157                                        | 3.57                                                 | 6.71                   | 0.23                        | 0.035                         | 171.9          | 297.7          | 148.8          |
| $Y_p$                  | 43.4              | 130.517                                       | 90.79                                                | 170.8                  | 5.96                        | 0.093                         | 64.7           | 112.1          | 56.1           |
| $Z_p$                  | 47.4              | 126.949                                       | 94.36                                                | 177.5                  | 6.20                        | 0.096                         | 62.6           | 108.5          | 54.2           |

Conclusions: GG stable orientation is:  $X_p$  - VERT,  $Y_p$  - POP,  $Z_p$  - AVV

With Orbiter

|       |      |         |        |        |      |       |      |       |      |
|-------|------|---------|--------|--------|------|-------|------|-------|------|
| $X_p$ | 21.3 | 71.605  | 32.31  | 60.79  | 2.12 | 0.075 | 80.4 | 139.2 | 69.6 |
| $Y_p$ | 16.8 | 144.700 | 105.41 | 198.3  | 6.92 | 0.095 | 63.3 | 109.6 | 54.8 |
| $Z_p$ | 19.8 | 177.012 | 73.10  | 137.51 | 4.80 | 0.071 | 84.0 | 145.5 | 72.8 |

Conclusions: GG stable orientation is:  $X_p$  - VERT,  $Y_p$  - AVV,  $Z_p$  - POP

\*Given as in-plane (pitch) characteristics

Note: Orbit altitude = 450 km

AVV = Along velocity vector

POP = Perpendicular to orbit plane

VERT = Along the vertical

Table 5-7

CONFIGURATION: SOLAR BRAYTON +100 MRAD POWER UNITS ROTATED 90°

| Axis                   | Rotation<br>(deg) | $I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | $\Delta I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | $T_{max}^*$<br>(ft-lb) | $T/\theta^*$<br>(ft-lb/deg) | $\dot{\theta}^*$<br>(deg/sec) | $P_p$<br>(min) | $P_y$<br>(min) | $P_r$<br>(min) |
|------------------------|-------------------|-----------------------------------------------|------------------------------------------------------|------------------------|-----------------------------|-------------------------------|----------------|----------------|----------------|
| <u>Without Orbiter</u> |                   |                                               |                                                      |                        |                             |                               |                |                |                |
| $X_p$                  | 20.2              | 35.658                                        | 9.43                                                 | 17.73                  | 0.62                        | 0.057                         | 105.0          | 181.9          | 90.9           |
| $Y_p$                  | 11.0              | 133.702                                       | 88.62                                                | 166.72                 | 5.82                        | 0.090                         | 66.3           | 114.9          | 57.4           |
| $Z_p$                  | 23.0              | 124.277                                       | 98.04                                                | 154.45                 | 6.44                        | 0.099                         | 60.5           | 105.3          | 52.6           |

Conclusions: GG stable orientation is:  $X_p$  - VERT,  $Y_p$  - POP,  $Z_p$  - AVVWith Orbiter

|       |      |         |        |        |      |       |      |       |      |
|-------|------|---------|--------|--------|------|-------|------|-------|------|
| $X_p$ | 21.3 | 71.535  | 24.60  | 46.28  | 1.62 | 0.065 | 92.1 | 159.5 | 79.7 |
| $Y_p$ | 18.0 | 148.599 | 101.66 | 191.26 | 6.68 | 0.092 | 65.3 | 113.1 | 56.5 |
| $Z_p$ | 22.5 | 173.198 | 77.06  | 144.98 | 5.06 | 0.074 | 80.9 | 140.2 | 70.1 |

Conclusions: GG stable orientation is:  $X_p$  - VERT,  $Y_p$  - AVV,  $Z_p$  - POP

\*Given as in-plane (pitch) characteristics

Note: Orbit altitude = 450 km

AVV = Along velocity vector

POP = Perpendicular to orbit plane

VERT = Along the vertical

Table 5-8

CONFIGURATION: SOLAR BRAYTON 90° ROTATION, SBU'S LOOK ALONG +Y

| Axis                   | Rotation<br>(deg) | $I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | $\Delta I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | Tmax*<br>(ft-lb) | T/ $\theta$ *<br>(ft-lb/deg) | $\dot{\theta}$ *<br>(deg/sec) | P <sub>p</sub><br>(min) | P <sub>y</sub><br>(min) | P <sub>r</sub><br>(min) |
|------------------------|-------------------|-----------------------------------------------|------------------------------------------------------|------------------|------------------------------|-------------------------------|-------------------------|-------------------------|-------------------------|
| <u>Without Orbiter</u> |                   |                                               |                                                      |                  |                              |                               |                         |                         |                         |
| X <sub>p</sub>         | 9.3               | 8.691                                         | 1.41                                                 | 2.65             | 0.09                         | 0.045                         | 134.2                   | 232.4                   | 116.2                   |
| Y <sub>p</sub>         | 26.9              | 26.551                                        | 16.45                                                | 30.95            | 1.08                         | 0.087                         | 68.6                    | 118.8                   | 59.4                    |
| Z <sub>p</sub>         | 25.7              | 25.144                                        | 17.86                                                | 33.60            | 1.17                         | 0.094                         | 64.1                    | 111.0                   | 55.5                    |

Conclusions: GG stable orientation is: X<sub>p</sub> - VERT, Y<sub>p</sub> - POP, Z<sub>p</sub> - AVVWith Orbiter

|                |      |        |       |       |      |       |      |       |      |
|----------------|------|--------|-------|-------|------|-------|------|-------|------|
| X <sub>p</sub> | 39.1 | 46.306 | 37.16 | 69.90 | 2.44 | 0.100 | 60.3 | 104.4 | 52.2 |
| Y <sub>p</sub> | 43.8 | 26.597 | 17.45 | 32.83 | 1.15 | 0.090 | 66.7 | 115.4 | 57.7 |
| Z <sub>p</sub> | 29.2 | 63.755 | 19.71 | 37.08 | 1.29 | 0.062 | 97.1 | 168.2 | 89.1 |

Conclusions: GG stable orientation is: X<sub>p</sub> - AVV, Y<sub>p</sub> - VERT, Z<sub>p</sub> - POP

\*Given as in-plane (pitch) characteristics

Note: Orbit altitude = 450 km

AVV = Along velocity vector

POP = Perpendicular to orbit plane

VERT = Along the vertical

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Table 5-9

CONFIGURATION: SOLAR BRAYTON 90° ROTATION, SBU'S LOOK ALONG -Y

| Axis                   | Rotation<br>(deg) | $I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | $\Delta I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | Tmax*<br>(ft-lb) | T/θ*<br>(ft-lb/deg) | $\dot{\theta}$ *<br>(deg/sec) | P <sub>p</sub><br>(min) | P <sub>y</sub><br>(min) | P <sub>r</sub><br>(min) |
|------------------------|-------------------|-----------------------------------------------|------------------------------------------------------|------------------|---------------------|-------------------------------|-------------------------|-------------------------|-------------------------|
| <u>Without Orbiter</u> |                   |                                               |                                                      |                  |                     |                               |                         |                         |                         |
| X <sub>p</sub>         | 12.4              | 8.822                                         | 1.44                                                 | 2.71             | 0.09                | 0.045                         | 133.7                   | 231.6                   | 115.8                   |
| Y <sub>p</sub>         | 31.5              | 27.003                                        | 16.74                                                | 31.50            | 1.10                | 0.088                         | 68.6                    | 118.8                   | 59.4                    |
| Z <sub>p</sub>         | 28.9              | 25.565                                        | 18.18                                                | 34.20            | 1.19                | 0.094                         | 64.0                    | 110.9                   | 55.4                    |

Conclusions: GG stable orientation is: X<sub>p</sub> - VERT, Y<sub>p</sub> - POP, Z<sub>p</sub> - AVV

With Orbiter

|                |      |        |       |       |      |       |      |       |      |
|----------------|------|--------|-------|-------|------|-------|------|-------|------|
| X <sub>p</sub> | 38.1 | 48.637 | 39.45 | 74.22 | 2.59 | 0.100 | 59.9 | 103.8 | 51.9 |
| Y <sub>p</sub> | 42.1 | 26.152 | 16.97 | 31.92 | 1.11 | 0.090 | 67.0 | 116.1 | 58.0 |
| Z <sub>p</sub> | 29.1 | 65.604 | 22.49 | 42.30 | 1.48 | 0.065 | 92.2 | 159.7 | 79.9 |

Conclusions: GG stable orientation is: X<sub>p</sub> - AVV, Y<sub>p</sub> - VERT, Z<sub>p</sub> - POP

\*Given as in-plane (pitch) characteristics

Note: Orbit altitude = 450 km

AVV = Along velocity vector

POP = Perpendicular to orbit plane

VERT = Along the vertical

Table 5-10

## CONFIGURATION: SOLAR BRAYTON SBU's &amp; MASTS AS CONCENTRATED MASS

| Axis                   | Rotation<br>(deg) | $I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | $\Delta I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | Tmax*<br>(ft-lb) | T/ $\theta$ *<br>(ft-lb/deg) | $\dot{\theta}$ *<br>(deg/sec) | P <sub>p</sub><br>(min) | P <sub>y</sub><br>(min) | P <sub>r</sub><br>(min) |
|------------------------|-------------------|-----------------------------------------------|------------------------------------------------------|------------------|------------------------------|-------------------------------|-------------------------|-------------------------|-------------------------|
| <u>Without Orbiter</u> |                   |                                               |                                                      |                  |                              |                               |                         |                         |                         |
| X <sub>p</sub>         | 8.6               | 4.028                                         | 3.73                                                 | 7.01             | 0.25                         | 0.107                         | 56.1                    | 97.2                    | 48.6                    |
| Y <sub>p</sub>         | 10.8              | 21.461                                        | 21.16                                                | 39.80            | 1.39                         | 0.110                         | 54.4                    | 94.2                    | 47.1                    |
| Z <sub>p</sub>         | 9.4               | 25.186                                        | 17.43                                                | 32.80            | 1.15                         | 0.092                         | 64.9                    | 112.4                   | 56.2                    |

Conclusions: GG stable orientation is: X<sub>p</sub> - VERT, Y<sub>p</sub> - AVV, Z<sub>p</sub> - POP

With Orbiter

|                |      |        |       |       |      |       |      |       |      |
|----------------|------|--------|-------|-------|------|-------|------|-------|------|
| X <sub>p</sub> | 39.7 | 41.853 | 40.04 | 75.33 | 2.63 | 0.109 | 55.2 | 95.6  | 47.8 |
| Y <sub>p</sub> | 43.6 | 22.308 | 20.50 | 38.56 | 1.35 | 0.107 | 56.3 | 97.5  | 48.8 |
| Z <sub>p</sub> | 24.9 | 62.351 | 19.55 | 36.77 | 1.28 | 0.062 | 96.4 | 167.0 | 83.5 |

Conclusions: GG stable orientation is: X<sub>p</sub> - AVV, Y<sub>p</sub> - VERT, Z<sub>p</sub> - POP

\*Given as in-plane (pitch) characteristics

Note: Orbit altitude = 450 km

AVV = Along velocity vector

POP = Perpendicular to orbit plane

VERT = Along the vertical

Table 5-11

CONFIGURATION: SOLAR BRAYTON +100 MRAD SBU'S & MASTS AS CONCENTRATED MASS

| Axis                   | Rotation<br>(deg) | $I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | $\Delta I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | Tmax*<br>(ft-lb) | T/ $\theta$ *<br>(ft-lb/deg) | $\dot{\theta}$ *<br>(deg/sec) | P <sub>p</sub><br>(min) | P <sub>y</sub><br>(min) | P <sub>r</sub><br>(min) |
|------------------------|-------------------|-----------------------------------------------|------------------------------------------------------|------------------|------------------------------|-------------------------------|-------------------------|-------------------------|-------------------------|
| <u>Without Orbiter</u> |                   |                                               |                                                      |                  |                              |                               |                         |                         |                         |
| X <sub>p</sub>         | 19.1              | 31.365                                        | 6.08                                                 | 11.43            | 0.40                         | 0.049                         | 122.7                   | 212.5                   | 106.2                   |
| Y <sub>p</sub>         | 17.3              | 130.717                                       | 93.28                                                | 175.48           | 6.13                         | 0.094                         | 63.9                    | 110.7                   | 55.3                    |
| Z <sub>p</sub>         | 25.9              | 124.642                                       | 99.35                                                | 186.91           | 6.52                         | 0.099                         | 60.5                    | 104.7                   | 52.4                    |

Conclusions: GG stable orientation is: X<sub>p</sub> - VERT, Y<sub>p</sub> - POP, Z<sub>p</sub> - AVV

With Orbiter

|                |      |         |        |        |      |       |      |       |      |
|----------------|------|---------|--------|--------|------|-------|------|-------|------|
| X <sub>p</sub> | 20.8 | 66.214  | 27.20  | 51.17  | 1.79 | 0.071 | 84.2 | 145.9 | 73.0 |
| Y <sub>p</sub> | 17.0 | 145.756 | 106.74 | 200.80 | 7.01 | 0.095 | 63.1 | 109.3 | 54.6 |
| Z <sub>p</sub> | 20.5 | 172.953 | 79.54  | 149.64 | 5.22 | 0.075 | 79.6 | 137.9 | 68.9 |

Conclusions: GG stable orientation is: X<sub>p</sub> - VERT, Y<sub>p</sub> - AVV, Z<sub>p</sub> - POP

\*Given as in-plane (pitch) characteristics

Note: Orbit altitude = 450 km

AVV = Along velocity vector

POP = Perpendicular to orbit plane

VERT = Along the vertical



Table 5-12

CONFIGURATION: SOLAR BRAYTON 2 SBU'S AND MASTS ALONE 90° ROT, LOOK ALONG -Y

| Axis            | Rotation<br>(deg) | $I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | $\Delta I \times 10^{-6}$<br>(slug-ft <sup>2</sup> ) | Tmax*<br>(ft-lb) | T/ $\theta$ *<br>(ft-lb/deg) | $\dot{\theta}$ *<br>(deg/sec) | P <sub>p</sub><br>(min) | P <sub>y</sub><br>(min) | P <sub>r</sub><br>(min) |
|-----------------|-------------------|-----------------------------------------------|------------------------------------------------------|------------------|------------------------------|-------------------------------|-------------------------|-------------------------|-------------------------|
| Without Orbiter |                   |                                               |                                                      |                  |                              |                               |                         |                         |                         |
| X <sub>p</sub>  | 0.0               | 4.690                                         | 4.50                                                 | 8.46             | 0.30                         | 0.109                         | 55.1                    | 95.5                    | 47.7                    |
| Y <sub>p</sub>  | 0.0               | 4.639                                         | 4.55                                                 | 8.56             | 0.30                         | 0.110                         | 54.5                    | 94.2                    | 47.2                    |
| Z <sub>p</sub>  | 0.0               | 0.140                                         | 0.05                                                 | 0.10             | 0.003                        | 0.067                         | 89.5                    | 154.9                   | 77.5                    |

Conclusions: GG stable orientation is: X<sub>p</sub> - POP, Y<sub>p</sub> - AVV, Z<sub>p</sub> - VERT

\*Given as in-plane (pitch) characteristics

Note: Orbit altitude = 450 km

AVV = Along velocity vector

POP = Perpendicular to orbit plane

VERT = Along the vertical

through an orbit on the sunlit side with the SBUs rotated 90 degrees about the second axis at the "sunrise/sunset" conditions is presented in Tables 5-5, 5-8, and 5-9.

A complete solution to the pointing and stabilization problem will take considerably more analytical effort; however, some insight into the problem is presented in Tables 5-10 through 5-12. In Tables 5-10 and 5-11, it is assumed that the SCB (with and without the 100-meter radiometer) has a point mass located at the center of gravity of the solar brayton units combination. In this manner, the desired orientation for the SCB can be determined independent of the required orientation of the solar brayton units. The additional torques necessary for the SBU's can be derived from Table 5-12, which gives the gravity gradient data for the solar brayton unit combination by itself. The maximum torques of  $\sim 8.5$  ft-lb about the  $x_p$  - and  $y_p$  -axes and  $\sim 0.10$  ft-lb about the  $z_p$  -axis were used to calculate momentum storage and impulse requirements, assuming that the remainder of the SCB is three-axis gravity gradient stabilized.

The effect of aerodynamic forces on these configurations was evaluated. Without mission hardware, the steady aerodynamic torque (assuming a high solar flux year, 1991, with an index of  $S_{10.7} = 170$ ) provides an offset angle of 1.4 degrees from the gravity gradient null. Because of the cyclic influence of the diurnal bulge of the atmosphere during the course of an orbit, the trim angle will actually vary from 0.9 degrees to 1.9 degrees. With the addition of the 100-meter radiometer, the trim angle becomes 2.6 degrees plus or minus 0.9 degrees. The addition of the Orbiter has little effect on these values.

Orbit-keeping and attitude-control propellant were determined for the configuration with and without the 100-meter radiometer. All the data associated with Orbit maintenance are presented in Table 5-13. The presence of the large radiometer has a dominating effect on these data because its large drag area affects the orbit-keeping propellant requirements as well as the projected lifetime and orbit-keeping schedule. Scheduled orbit-keeping permits the orbit altitude to decay from a constant altitude (and continuous thrust) condition and incurs a 5 percent propellant penalty.

The resource benefits of utilizing control moment gyro's (CMGs) as momentum storage devices for stabilizing the vehicle are shown in Table 5-13. The

Table 5-13  
SOLAR BRAYTON ORBIT-KEEPING REQUIREMENTS  
(450 km, no Orbiter)

|                                           | No Mission<br>Hardware | With 100-Meter<br>Radiometer |
|-------------------------------------------|------------------------|------------------------------|
| CdA (ft <sup>2</sup> )                    | 18,304                 | 85,474                       |
| W/CdA (lb/ft <sup>2</sup> )               | 8.75                   | 3.19                         |
| Lifetime* (weeks)                         | 33                     | 12                           |
| Orbit-Keeping Interval<br>(Days)**        | 17                     | 6                            |
| Orbit-Keeping<br>(Propellant (kg/mo)**    | 150                    | 702                          |
| Attitude Control*<br>Propellant (kg/mo)** | 25/215                 | 131/470                      |
| Total (kg/mo)                             | 175/365                | 833/1,172                    |

\*Assuming no orbit-keeping

\*\*5% interval factor penalty, I<sub>SP</sub> = 270 sec

\*\*\* ( )/( ) with and without CMGs

benefit of using CMGs is in stabilizing an unstable axis about null with minimum propellant expenditure and in absorbing cyclic torques due to solar brayton unit two-axis gimbaling. The net result significantly reduces the amount of propellant required, as indicated. Accordingly, the preferred system is one which utilizes CMGs. The momentum storage size appears to be on the order of 5000 ft-lb-sec, within the capability of the ATM system (6900 ft-lb-sec) that has flown on Skylab.

An approach to the vernier pointing system for the SBUs is outlined below.

A. Requirement for pointing

1. Small power losses up to 15 arc minutes error
2. Six percent power loss at 30 arc minutes error.

B. Gross pointing

1. Two-axis (large freedom) gimbaling
  - a. Spline extension from end of strongback

- b. Spline orthogonal to above
  2. Solve gravity gradient torques  $\approx 0$  as function of position in orbit, orbit inclination, and principal axis tilt
  3. Points solar brayton units toward sun to about 1 degree accuracy
  4. Fairly complex software, but solvable.
- C. Vernier pointing
1. Two-axis (limited freedom, about  $\pm 1$  degree) gimbaling about axes perpendicular to sun line of sight
  2. Accurate solar sensors mounted to periphery of collector (need to be baffled from radiator and engine room)
  3. If sensor alignment not adequate, can perform bias calibration.

It is expected that the gross pointing system accuracy will be of the order of 1 degree. To provide a pointing error of 15 arc minutes, each solar brayton unit will require two-axis limited-freedom gimbals and sun sensors to provide the basic pointing information. To test the alignment calibration of the sensors relative to the focus axis, small perturbations can be introduced and held for the duration of the sunlit portion of the orbit, and the total power output of each solar brayton unit can be monitored. This technique may take several orbits to seek the maximum power trim conditions, but, considering the long duration of the mission, it appears negligible, and affords a method for periodic alignment calibration.

### Summary

The summary analysis of the attitude and flight control of the SCB with solar brayton power is given below.

- Gravity gradient stabilization is recommended about two axes for long-term orientation.
- Stabilization about a third axis (vertical) and accommodation of gross pointing of solar brayton units suggests CMG momentum storage and actuation system.
- Vernier pointing system with limited freedom required to ensure 15 arc minutes accuracy.

Because of the complexity introduced by gross pointing of the solar brayton units, momentum storage in the form of CMG's is recommended. Two-axis

gravity gradient stabilization is recommended; however, the third axis, yaw (or azimuth), requires more angular freedom for the orbiting vehicle and the ability to absorb off-trim moments from the solar brayton unit gimbaling.

### 5.3.3 Thermal Analysis

Thermal analysis of the solar brayton power system addressed the effort to analyze, develop, and analytically verify a preliminary design which interfaces efficiently with other system equipment. The thermal equipment addressed in this section includes that which transfers heat from the receiver to the brayton cycle working fluid and that which removes heat from the working fluid for rejection overboard with radiators.

The thermal requirements, analytical approach, and study results are discussed in the following summary.

#### Summary

Two key thermal components are addressed in this section: the heat source heat exchanger and the heat sink radiator. These system elements were examined in detail and sized to accommodate the preliminary design of the solar brayton power system.

Previous brayton cycle engine systems have utilized heat transfer tubes located in the solar heat receiver to transfer the thermal energy from the receiver to the brayton cycle working fluid (Reference 5-3). The current investigation considers the use of heat pipes to transfer the thermal energy from the receiver to remote heat source heat exchangers. The use of heat pipes offers the following advantages: (1) reduction in the number of potential leakage paths for the cycle working fluid, (2) reduction in the total surface area of the working fluid containers (tubing, manifolds, etc.) exposed to potential meteorite penetration, (3) allows one solar heat receiver to be utilized with three separate brayton power plants with completely isolated working fluid systems, and (4) loss of a single heat pipe (e. g., due to meteorite penetration) would result in only a slight loss in input thermal energy to the working fluid, whereas failure of a single heat transfer tube (previous configurations) would result in the complete loss of the working fluid.

The radiator conceptual design was analyzed to assess performance capability and to obtain insight into design and performance drivers. Analysis showed that the current cruciform design has a performance margin of about 38% at the design case of beta angle equal to zero. Additionally, the analysis showed that the degradation of radiator surface coatings has a negligible effect on performance. Although the design case considered a beta angle equal to zero, the performance is expected to improve at higher beta angles due to the reduced incidence of earth albedo heat influx.

### Requirements

The functional requirement for the heat source heat exchanger is to transfer heat from the solar receiver via heat pipes to the heat source heat exchanger. Sufficient heat storage capability must exist to supply heat to the brayton cycle during shade portions of the orbit.

The current solar-powered brayton power plant has an electrical output of 60 kWe and requires a continuous thermal energy input to the working fluid of 223.2 kW (heat pipe heat exchanger heat load). As described in Reference 5-3, a maximum allowable pressure drop in the heat exchanger of 2 percent of the inlet pressure and a lithium fluoride heat storage system were utilized; therefore, the operating temperature of the heat pipes was 1,122 K (2,020°R), the heat exchanger inlet temperature (cycle working fluid) was 866 K (1,558°R), and the required outlet temperature was 1,088 K (1,960°R).

The heat rejection element of the design serves the purpose of removing heat at the heat sink portion of the brayton cycle and transporting and rejecting it to space. The radiator consists of a deployed, four element radiator which must reject 163 kW of heat. Inlet temperature to the radiator is 411 K (741°R) and the outlet temperature is 292 K (526°R). The configuration and dimensions of the radiator system were given earlier in Figure 5-9.

### Analysis

This section describes the assumptions, approach, and results of the analysis.

Radiator. The radiator configuration analyzed consisted of a flat plate deployed radiator as depicted in Figure 5-9. Dow Corning-200 is the circu-

lating fluid which passes through tubes on 15.2 to 20.3 cm (6 to 8 inch) centers. The aluminum radiator surface was assumed to be sufficiently thick, about 0.5 mm (0.02 inch), to obtain a fin effectiveness of at least 90 percent. Flow direction in the radiator is of minor importance since the heat flux does not vary greatly over the surfaces. Two-sided heat rejection was assumed.

The effects of the collector were neglected. In actual practice the collector should block earth IR and albedo on the sun side of the orbit, but will block cold outer space on the shade side for orientations placing the radiators between the earth and collector. It is believed that these two effects offset one another.

A solar orientation was assumed for the analysis. This is the only possible orientation on sun side; however, other orientations are possible on the shade side of the orbit. A solar orientation (inertial hold) was assumed also for shade side. A beta angle of zero was analyzed since this is expected to be the worst design case.

Although the effects of the collector were neglected, the interchange between the adjacent radiators was accounted for. Only direct radiation was considered; reradiation or reflected energy was neglected.

The technical approach was based on the exact solution analysis reported in Reference 5-4. This solution utilizes a flat plate with the environment represented by an effective sink temperature defined as follows:

$$T_s = \left\{ \frac{1}{2\sigma} \left[ \frac{\alpha_s}{\epsilon_t} SF_s + \frac{\alpha_s}{\epsilon_t} a SF_{E(SR)} + \frac{(1-a)}{4} SF_{ET} + \frac{2 F_R E_{R(IR)}}{\epsilon_t} \right] \right\}^{1/4} \quad (5-1)$$

where:

$\sigma$  = Stefan - Boltzmann constant

$T_s$  = sink temperature

$\alpha_s$  = absorptivity in solar wavelength

$S$  = solar constant  
 $F_s$  = view factor, solar  
 $a$  = earth albedo  
 $F_{E(SR)}$  = view factor, solar reflected  
 $F_{ET}$  = view factor, earth thermal  
 $F_R$  = view factor, adjacent radiator  
 $E_{R(IR)}$  = heat flux from adjacent radiator  
 $\epsilon_t$  = emissivity, thermal

Values for the various view factors were obtained from Reference 5-4 for different orbital locations and orientations.

The sink temperature approach allows a simple equation to be written for heat transfer from an increment of radiator area:

$$dQ_{R, \text{ net}} = \sigma \epsilon_t \left[ T^4 - T_s^4 \right] dA \quad (5-2)$$

where,

$T$  = temperature of increment  
 $A$  = radiator area

Upon setting heat transfer from the radiator increment equal to heat transferred from the radiator fluid in the increment and integrating the resulting equation, a closed-form solution can be obtained:

$$\xi(\tau_1) - \xi(\tau_2) = \frac{\sigma \epsilon_t \eta_o T_s^3 A_R}{wc_p} \quad (5-3)$$

where,

$\tau = \frac{T_R}{T_S} = \frac{\text{temperature radiator}}{\text{temperature sink}}$   
 $\eta_o$  = fin effectiveness



$A_R$  = radiator area  
 $W$  = fluid flow rate  
 $C_p$  = specific heat  
1 = subscript for inlet  
2 = subscript for outlet

$$\xi(\tau) = \frac{1}{4} \ln \left[ \frac{\tau+1}{\tau-1} \right] + \frac{1}{2} \tan^{-1} \tau \quad (5-4)$$

The above equations require an iterative solution wherein Equation (5-3) must be solved for  $\xi(\tau_2)$ ; all other terms are known. In simple terms, the solution must be found for  $\xi(\tau_2) = \text{constant}$ . This requires an iterative solution since  $\tau$ , Equation (5-4), cannot be solved in closed form.

Both heating and cooling conditions can be obtained by use of this solution; however, absolute values of the  $\ln$  term must be used in the heating case. Additionally, the following modified solution was used for cases where the sink temperature was zero:

$$T_{R2} = \left[ \frac{T_{R1}^3 W C_p}{3 \sigma \epsilon_t \eta_o A_T T_{R1}^3 + W C_p} \right]^{1/3} \quad (5-5)$$

As mentioned above, the solution is for steady state; however, a prediction of performance was obtained by the average performance for 12 points around the orbit. A constant radiator inlet temperature was used for all points in the orbit; the parameter  $A_R/WC_p$  was varied to obtain a range of radiator outlet temperatures; and heat rejected from the radiator was calculated based on radiator flow rate and temperature drop.

Results of the analysis are given in Figure 5-23 for the range of surface coating characteristics. The results show that for the required outlet temperature of 292 K (526°R), the radiator can reject from 0.429 to 0.445 kW/m<sup>2</sup> (136 to 141 Btu/hr/ft<sup>2</sup>), depending upon the degree of coating degradation. This performance is about 38 percent higher than is required for the 60 kWe power system.

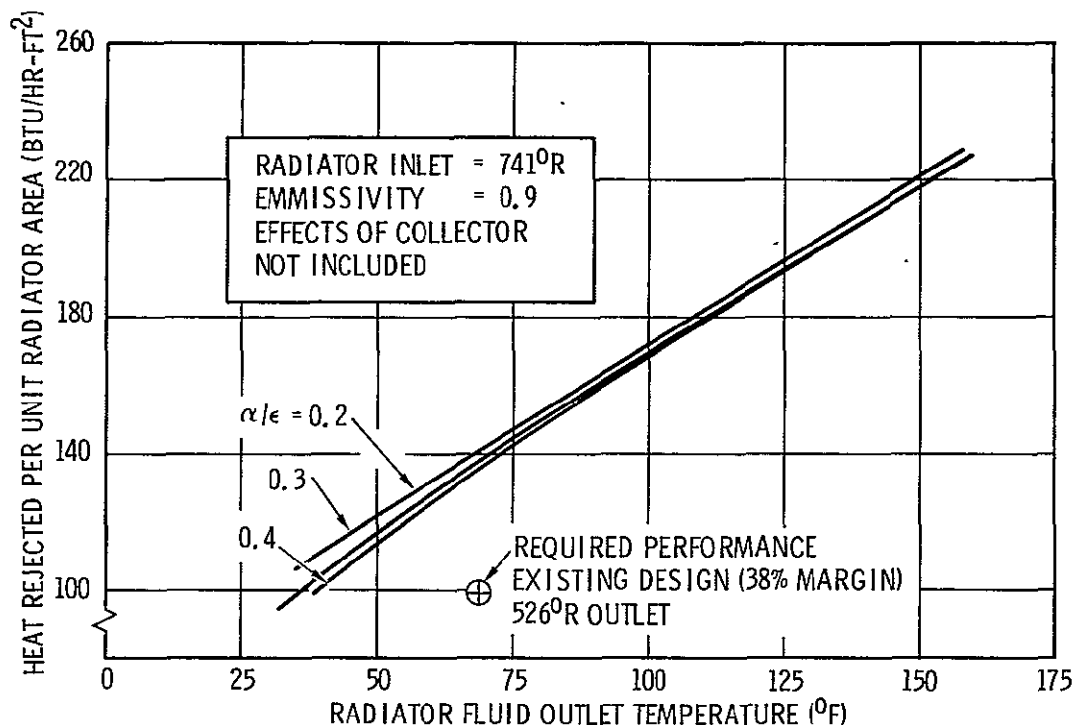


Figure 5-23. Solar Brayton Power System Radiator Performance

Heat Source Heat Exchanger. The conical wall of the solar heat receiver is formed by the evaporator ends of the heat pipes, with each heat pipe encapsulated by the heat storage tubes (Reference 5-2). The operating temperature of the lithium fluoride phase change material dictated the use of sodium as the heat pipe working fluid. Sodium has a typical operating range of 870 K (1,566°R) to 1,270 K (2,286°R) in heat pipe applications. The heat pipe container material selected was stainless steel.

The diameter and number of heat pipes to be utilized was dictated by the geometry of the receiver and the limiting axial heat flux for sodium heat pipes. Utilizing 3.2 cm (1.25 inch) outside diameter heat pipes, the receiver could accommodate 50 heat pipes with a total required heat rate of 223.2 kW. This would result in a heat pipe axial flux of  $7.42 \times 10^6 \text{ W/m}^2$  ( $2.35 \times 10^6 \text{ Btu/hr-ft}^2$ ). This flux is approximately an order of magnitude below the limiting fluxes for 100/200 mesh screen wicks.

The radiator ends of the heat pipes are finned, spirally wound, continuously welded at the base, and composed of stainless steel. The heat pipes are

bundled in a staggered, equilateral triangular arrangement (as viewed from the end). The fin geometry was selected as that being readily available from finned-tube manufacturers (maximum height, maximum number of fins/unit length, minimum thickness) and is 1.27 cm (0.5 inch) high, 0.64 mm (0.025 inch) thick, 2.76 fins/cm (7 fins/inch).

Based on the He/Xe working fluid flow rate of 4.03 kg/sec (8.9 pounds per second), the required heat rate of 223.2 kW, and estimated sodium heat pipe performance characteristics, the length of each heat exchanger (heat pipe radiator segment) was determined to be 0.76 meters (2.5 feet). The radiator portions of the heat pipes are divided longitudinally into three equal segments to permit the utilization of the solar heat receiver by three autonomous brayton power plants. The final heat exchanger layout was determined to be 11 rows of heat pipes (alternate rows of five and four), resulting in an 0.76 meter by 0.30 meter (2.5 feet by 1 foot) heat exchanger face area (normal to the He/Xe flow) and a depth of 0.61 meter (2 feet).

The pressure drop for the heat exchanger was determined to be 0.23 N/cm<sup>2</sup> (0.33 psi).

#### Discussion of Results

The radiator analysis showed that a 38 percent margin exists with the configuration analyzed. This will result in a colder heat sink than required for the brayton cycle, and will increase output somewhat. A second alternative would be to reduce the radiator size, thereby resulting in a small weight, volume, and power savings.

The analysis also showed that radiator surface characteristics have little effect on performance because direct solar radiation never impinges directly on the radiator and earth albedo effects are not great.

The view factors from adjacent radiators are relatively small, amounting to about 0.078. This verifies that the current cruciform radiator arrangement is efficient.

As mentioned in the previous section on analysis, the collector was neglected in the analysis because of the analytical complexity which would result. A

secondary analysis was performed to estimate the temperature of the collector at various points around the orbit. The variation was from 281 K (506°R) on the sun side to 117 K (211°R) on the shade side. These temperatures are much lower than radiator temperatures and this, combined with the low emissivity of the collector, should result in little direct heat influx. However, the collector will reflect heat energy very effectively from radiators and from earth IR and albedo. The view factors for these effects are small, thus were neglected for preliminary design type analyses.

Only a beta angle of zero was considered in the analysis since this represented the worst design case due to the high earth albedo heat influx for this condition. Performance is expected to improve at higher beta angles since earth albedo effects will decrease.

Each heat source exchanger was sized to meet the required brayton cycle heat load of 223.2 kW. The calculated pressure drop ( $0.23 \text{ N/cm}^2$ ) was well within the design requirement of 2 percent of inlet pressure: Based on an inlet pressure of  $37.21 \text{ N/cm}^2$  (54 psia), the allowable pressure drop is  $0.74 \text{ N/cm}^2$  (1.08 psi).

As previously discussed, the performance of each heat exchanger is relatively insensitive to the loss of a single heat pipe, and, in this case, would still perform at approximately 99 percent of normal operation.

#### 5.3.4 Reliability Analysis

A brief study was conducted to determine the reliability and maintenance characteristics of a solar brayton power system for use in a Space Station application.

The source of thermal energy in the solar brayton power system is solar radiation which is collected by a concentrator which has a diameter of 21 meters and a surface area of 375 square meters. The solar energy is focused by the concentrator into a cavity receiver, where the thermal energy is collected and transferred to the brayton unit. The electrical output of one solar brayton power system unit is 60 kWe.

### Brayton Power Unit

The reliability and maintenance characteristics of the brayton power unit were analyzed first because they are common to both power systems and were discussed earlier in Section 4.3.5. The results are identical.

### Solar Brayton System

The results of the analysis of the solar brayton system are shown in Table 5-14. A 10-year reliability of 0.990 was assigned to the dish con-

Table 5-14  
SOLAR BRAYTON POWER SYSTEM

| Item                | Operating Units | Basic Failure Rate ( $10^6$ Hr)<br>(10-Yr Reliability) | 10-Yr System Reliability |            |
|---------------------|-----------------|--------------------------------------------------------|--------------------------|------------|
|                     |                 |                                                        | No Maint                 | With Maint |
| Dish                | 1/1             | 0.9990                                                 | 0.9990                   | 0.9990     |
| Receiver            | 1/1             |                                                        | 0.8756                   | 0.8757     |
| Heat Pipes          | 40/50           | 0.1                                                    | 0.9358                   | 0.9358     |
| Thermal Storage     | 40/50           | 0.1                                                    | 0.9358                   | 0.9358     |
| Thermal Doors       | 1/2, 2/4        | 1.0                                                    | 0.9999                   | 1.0000     |
| Heat Exchanger      | 1/2             | 0.9080                                                 | 0.9915                   | 0.9915     |
| Power Conversion    | 1/3             | 0.5435                                                 | 0.9245                   | 0.9936     |
| Radiator System     | 1/2             | 0.9411                                                 | 0.9965                   | 0.9965     |
| Radiator            | 1/1             | 0.9484                                                 |                          |            |
| Pumps               | 1/2             | 1.0                                                    |                          |            |
| Instrumentation     | 100 sets        |                                                        | 0.9998                   | 0.9999     |
| Sensors             | 2/5             | 0.1                                                    | 0.9999                   | 0.9999     |
| Signal Conditioning | 2/5             | 0.1                                                    | 0.9999                   | 1.0000     |
| System              |                 |                                                        | 0.7989                   | 0.8588     |

centrator. The receiver is a cavity receiver where the solar rays enter the cavity through an aperture. There are four emergency heat-dump doors and only two are required for operation. Each door has two actuators; therefore, there is a 2/4 logic on the door system and a 1/2 logic on each door. The only failure-causing component is the actuator with a failure rate of 1.0 failures per million hours. (The emergency heat dump allows dissipation of excess solar input during high-beta-angle periods in the orbit where eclipse time is decreased.)

The thermal energy is transmitted from the cavity receiver via 50 heat pipes to an intermediate heat exchanger. It was assumed that a loss of 20 percent of the heat pipes would constitute a failure. A set of 50 thermal storage tubes is utilized to store thermal energy to provide an output for short periods when the collector is not illuminated. It was assumed that a loss of 20 percent of these would constitute a failure.

The 10-year reliability with maintenance is 0.8588 (Table 5-14).

The analysis of the intermediate heat exchanger, the power conversion system, the radiator system, and the instrumentation are the same as discussed in Section 4.3.5.

### Results

The output of the solar brayton power system is 60 kWe as opposed to the 120 kWe electrical power output of the reactor system. Therefore, two cases are shown in Table 5-15, but only the maintenance case is used. The results show that a 2/4 logic (at least two out of four units will operate properly) is required to approach the 0.999 level if 120 kWe are required, but only a 1/3 logic (total of 3 units) is required if only 60 kWe are required.

Table 5-15

10-YEAR RELIABILITY, SOLAR BRAYTON UNIT WITH MAINTENANCE

| 120 kWe |             | 60 kWe |             |
|---------|-------------|--------|-------------|
| Units*  | Reliability | Units* | Reliability |
| 2/2     | 0.7375      | 1/1    | 0.8588      |
| 2/3     | 0.9768      | 1/2    | 0.9884      |
| 2/4     | 0.9982      | 1/3    | 0.9991      |

\*Assumes standby redundancy

### Maintenance Analysis

The maintenance analysis was conducted by determining the number of failures expected, on a probabilistic basis, for each of the maintainable items (Table 5-16). In addition, an estimate of the average time to accomplish the repair or replacement is made. As shown in Table 5-16, the time

Table 5-16  
MAINTENANCE ANALYSIS

| Item                  | Solar System |              |
|-----------------------|--------------|--------------|
|                       | Fail/10 Yr   | Hr/10 Yr     |
| Power Conversion      |              |              |
| Controls              | 0.53         | 2.23         |
| Gas Management        | 0.42         | 1.76         |
| Instrumentation       |              |              |
| Signal Conditioning   | 4.38         | 18.4         |
| Solar Power Plant     |              |              |
| Thermal Doors         | 0.78         | 2.94         |
| Solar Pointing System | 1.04         | 6.03         |
| Unit Totals           | <u>7.15</u>  | <u>31.36</u> |
| 120 kW System Totals  | 14.30        | 62.72        |
| *Wait Time            | 2.5 Hr       |              |
| Detect                | 0.2          |              |
| R/R                   | 1.0          |              |
| C/O Adj               | <u>0.5</u>   |              |
| Total                 | 4.2 Hr       |              |

Table 5-17  
SOLAR BRAYTON POINTING SYSTEM

| Item                              | Basic<br>Failure<br>Rate<br>(10 <sup>6</sup> Hr) | Failure<br>Per<br>10 Yr | Maintenance<br>Hour/<br>Failure | Maintenance<br>Hours/<br>10 Yr |
|-----------------------------------|--------------------------------------------------|-------------------------|---------------------------------|--------------------------------|
| Motor                             | 3.0                                              | 0.263                   | 4.2                             | 1.10                           |
| Gear                              | 1.0                                              | 0.088                   | 6.2                             | 0.54                           |
| Chain Drive                       | 0.5                                              | 0.044                   | 4.2                             | 0.18                           |
| Bearing                           | 1.8                                              | 0.158                   | 9.2                             | 1.45                           |
| Seal                              | 0.40                                             | 0.035                   | 9.2                             | 0.32                           |
| Total                             |                                                  | 0.588                   | —                               | 3.59                           |
| Three Units                       |                                                  | 1.763                   | —                               | 10.77                          |
| Electronics                       | 3.5                                              | 0.307                   | 4.2                             | 1.29                           |
| Total System<br>(Two Power Units) |                                                  | 2.070                   | —                               | 12.06                          |

allocation is made up of four separate estimates: a wait time defined as time spent in putting on and taking off a space suit, the time used in detecting which component failed, the actual time spent repairing or replacing the component, and the time involved in checkout or adjustment of the repair or replaced component. The maintenance hours per 10 years per system is then the product of the number of failures and the maintenance hours per failure. On this basis each solar brayton unit would be expected to have about seven failures in 10 years and require 31 hours for maintenance. If we require 120 kWe of electrical power from the solar units, number of failures and repair time will double. One of the items in the maintenance analysis of the solar unit is the pointing system for the solar concentrator, which is shown in Figure 5-10. Three motor-gear-chain drive-bearing-seal units are involved. One unit turns both solar units around the axis of the Space Station, while the other two move the two solar units around an axis 90 degrees to the x-axis of the Space Station. The detailed maintenance analysis of the pointer system indicates a total of two failures per 10 years and 12 hours of maintenance time (Table 5-17).

#### Reliability Optimization

A cost effective reliability goal should be established in the manner described in Section 4.3.5.

#### 5.3.5 Micrometeoroid Damage Analysis

The micrometeoroid environment in low earth orbit is of concern for its effect on the mirror surface of the solar concentrator, i.e., surface degradation from impact would degrade the system power output.

Early analysis (1964) of micrometeoroid degradation is shown in Figure 5-24. The geocentric flux model, shown by the dash lines in the left hand curve, was used by Loeffler, Clough, and Lieblin to obtain the time-dependent degradation shown in the right-hand curve. As indicated, the falloff in surface reflectivity was calculated to be about 35 percent after one year in LEO. The currently accepted near earth flux shown in the left-hand curve was defined by NASA SP-8013 in 1969. Recent analysis of meteoroid technology satellite (MTS) data by Alvarez has extended the SP-8013 flux determination in the particle size region from  $10^{-11}$  to  $10^{-16}$  grams. In the



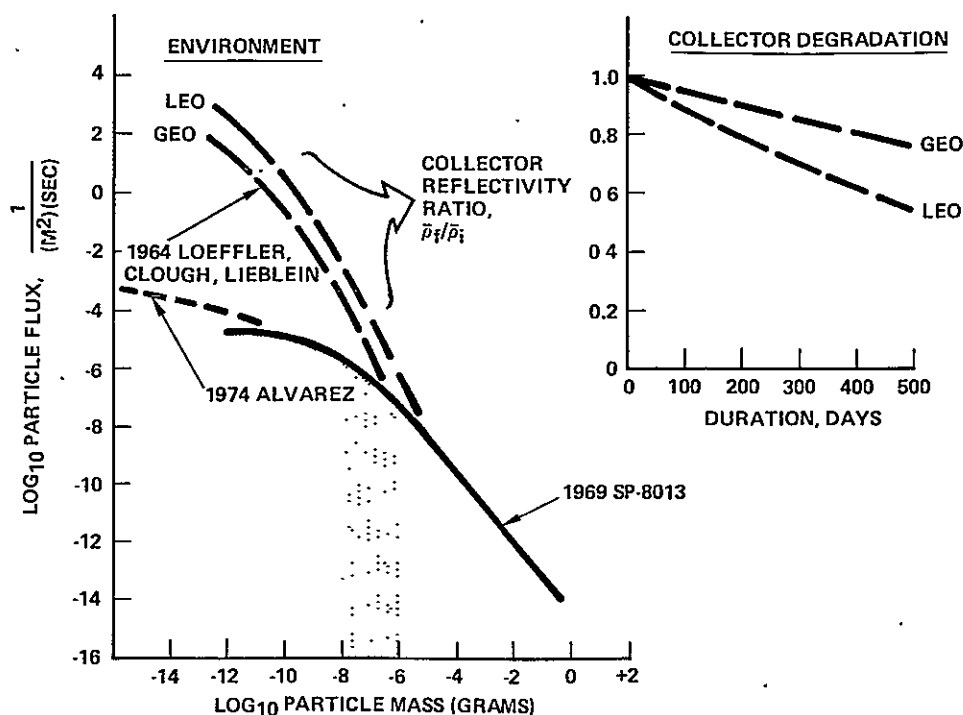


Figure 5-24. Micrometeoroid Degradation of Collector Surface

region between  $10^{-8}$  and  $10^{-12}$  gram particles, there is a difference of three to seven orders of magnitude between current data and that from 1964.

A calculation was made using the combined SP-8013 and Alvarez MTS data to develop the areal flux which could impact the concentrator surface in a one-year time period. A particle density of  $0.5 \text{ gram/cm}^3$  was used in accordance with SP-8013. This integrated areal flux represents the equivalent percentage of the projected collector area which would be swept by the LEO flux (Figure 5-25). Small particles ( $\leq 10^{-12}$  grams), although large in number, have minimal areal contribution. The predominant impact is from particles in the  $10^{-7}$  to  $10^{-9}$  gram region with a total area effect of about 1.7 percent per year. Actual surface degradation should be a function of particle size, impact angle, etc. However, it appears that micrometeoroid degradation should be significantly less than earlier predictions, and it is suggested that detailed analysis be accomplished to reduce uncertainties about the degradation mechanisms and their degree.

#### 5.4 SUMMARY AND CONCLUSIONS

Following are summary observations made after analysis of the solar brayton

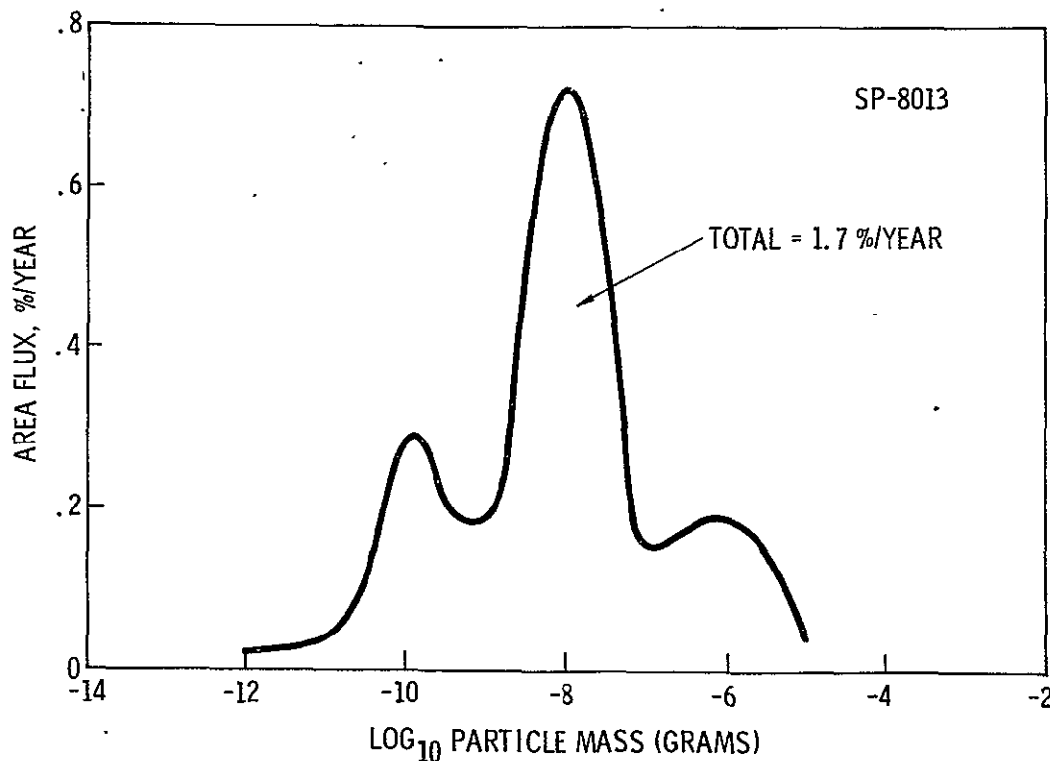


Figure 5-25. Predicted Concentrator Degradation

power system as influenced by preliminary integration with the SCB program.

- The 60-kWe power module design accommodates both initial and growth SCB power requirements by operation of either one or two modules. A potential exists for significant power level increase (up to 50 percent) by optimization of the brayton engine design while holding the concentrator size constant. Power growth options are given below.
  - A. Eliminate Orbiter docking module.
  - B. Assess cycle  $\eta$  gains at current temperatures.
  - C. Assess cycle  $\eta$  gains at higher temperatures and alternate to lithium fluoride.
  - D. Add modules.
  - E. Launch mirror segments separately.
- The solar brayton power system is not limited to launch and assembly with an existing SCB, but is acceptable as a free-flyer assembled in orbit by the shuttle. Power system initial launch costs for SCB buildup and for shuttle erected modes are given in Table 5-18. The incremental cost of the free-flyer approach should be examined in

Table 5-18  
SOLAR BRAYTON BASELINE AND OPTION

|                                     | Single Launch (Baseline)       |        | Free-Flyer<br>(Shuttle-Erected)          |        |
|-------------------------------------|--------------------------------|--------|------------------------------------------|--------|
| First Launch                        | Power module                   |        | Power module                             |        |
|                                     | Boom/gimbals/mast              |        | Boom/gimbals/mast                        |        |
|                                     |                                |        | Docking adapter                          |        |
|                                     |                                |        | Free-flight package                      |        |
| Second Launch                       | Power module, mast/<br>adapter |        | Shuttle docking module,<br>assembly aids |        |
| Third Launch                        | N/A                            |        | Power module, mast/<br>adapter           |        |
| Cost (in<br>millions of<br>dollars) | Basic solar<br>brayton pkgs    | ~\$107 | Basic solar<br>brayton pkgs              | ~\$107 |
|                                     | Integration pkgs               | 16     | Integration pkgs                         | 16     |
|                                     | Launches (2)                   | 40     | Free-flight pkg                          | 18     |
|                                     |                                |        | Launches (3)                             | 60     |
|                                     |                                |        | Shuttle pwr aug                          | 5      |
|                                     | Total                          | ~\$163 | Total                                    | ~\$206 |

light of the \$175 million cost for the 38 kWe power module which is not required in the free-flyer case.

- Provisions for maintainability are endorsed by the sponsoring NASA agency, and the designed reliability is equivalent to that for the reactor brayton power system. Determination of a cost-effective reliability goal for the system is suggested when system definition warrants the effort.
- Development risks for the solar brayton power system are summarized below. The main areas of concern arise from space environment effects on the collector and from the need for tight pointing accuracy. Analysis and tests are needed as indicated.
  - A. Solar concentrator
    1. Assessment of micrometeoroid flux effects
    2. Assessment of contamination effects, Orbiter and SCB
    3. Assessment of mirror resurfacing/replacement need

4. Boresighting of concentrators and retention of fine pointing micro gimbal adjustment.
- B. Receiver/heat storage element -- long-term behavior of lithium fluoride with sodium heat pipe.
- C. Receiver to brayton heat exchanger interface
  1. Materials-temperature compatibility
  2. Capability for redundancy (failure probability) - complexity.
- In view of these development risks, a relative system development risk of low to medium appears appropriate for system research and technology requirements. These requirements include:
  - A. Concentrator assembly/petal alignment
    1. Initial boresighting of two modules with common orbit rate
    2. Effects of temperature cycling/shadowing.
  - B. Concentrator surface degradation data
    1. Contamination from SCB/Orbiter effluents
    2. Micrometeoroid damage
    3. Potential for space resurfacing and replacement.
  - C. Design definition for concentrator pointing system.
  - D. Definition of thermal storage capability vs altitude and inclination
    1. Assessment of heat dump doors vs heat pipe—heat dump
    2. Assessment of heat pipe material compatibility with LiF1
    3. Appropriate thermal reserve.
  - E. Layout and packaging of a 60 kWe brayton engine with all heat exchanger elements and ancillary equipment

Section 6  
SOLAR ARRAY SYSTEMS

This section summarizes the baseline SCB solar array system and alternative solar array system options. The prime solar array system options considered are presented in Table 6-1, which indicates the two basic options of solar arrays that are fabricated and assembled and arrays that are deployed (Shuttle/Free-Flyer mode). The energy storage and silicon solar cell options are listed on the left of the table. Point designs were conducted for the combinations denoted by X (baseline), and extrapolated to provide comparative data for the combinations denoted by  $\Delta$  (delta). The NiCd battery and current SEP options represent low-risk, relatively current technology, appropriate for a conservative SCB program with a mid-1985 IOC (1980-proven technology). The other candidates are higher-risk, advanced technology options that show considerable promise for improvement in cost and performance.

Table 6-1  
POWER PLATFORM SOLAR ARRAY SYSTEM OPTIONS

| Buildup/Support Mode ➡                       | Fabrication and Assembly            |                        | Deployed-Shuttle/<br>Free-Flyer     |
|----------------------------------------------|-------------------------------------|------------------------|-------------------------------------|
|                                              | Power<br>Module                     | Shuttle/<br>Free-Flyer |                                     |
| Energy Storage                               |                                     |                        |                                     |
| ● NiCd Battery                               | <input checked="" type="checkbox"/> | Δ                      |                                     |
| ● Advanced NiCd Battery                      | Δ                                   |                        |                                     |
| ● NiH <sub>2</sub> Battery                   | Δ                                   |                        |                                     |
| ● Regenerative Fuel Cell                     | Δ                                   |                        | <input checked="" type="checkbox"/> |
| Solar Cells                                  |                                     |                        |                                     |
| ● Current SEP (11% η )                       | <input checked="" type="checkbox"/> | Δ                      | <input checked="" type="checkbox"/> |
| ● High Efficiency (14% η )                   |                                     |                        | Δ                                   |
| ● Low Cost (12% η )                          |                                     |                        | Δ                                   |
| <input checked="" type="checkbox"/> BASELINE |                                     |                        |                                     |

## 6.1 BASELINE SYSTEM DESCRIPTION – POWER PLATFORM (FABRICATION AND ASSEMBLY ARRAY)

Figure 6-1 shows an overview of the baseline SCB; the basic sizing and construction approach was accomplished during the basic SSSAS study and is summarized here to aid the power system comparison task. The SCB power platform features a solar array that is fabricated and assembled in orbit. The solar array is gimballed  $\pm 180$  degrees about the X axis, as noted. The average power output is 100 kWe (beginning-of-life – BOL) under typical orbit conditions ( $\beta = 23$  degrees and illumination =  $1353 \text{ W/m}^2$ ), as noted in Figure 6-2. The array output for these conditions is 214.7 kWe. Figure 6-2 details the baseline SCB power platform system, which stores energy in current-technology NiCd batteries. These batteries, the battery chargers, and the dc/dc output regulators of the energy storage subsystem are located along the bottom of the U-shaped channel that serves as both an assembly fixture and power system radiator. The radiator consists of the two sides of the U-shaped channel ( $96 \text{ m}^2$ ) plus a small, deployable radiator of  $64 \text{ m}^2$ . The  $\pm 180$ -degree gimbal employs a trailing cable, which is unwound during the eclipse portion of the orbit.

Space-fabricated triangular 1.0-m-wide beams of composite material are used for the 104.3-m-long solar array structure. The 8-m-wide solar cell blanket is attached to the longerons and cross beams by a system of negator springs. The solar cells are located on the upper side, as noted. The total active area of the solar cell blanket is  $8 \times 3 \times (104.3\text{m} - 4.0\text{m}) \text{ m} = 2407\text{m}^2$ , allowing for blockage by the four 1-m-wide cross members. The design and fabrication of the blanket were based on current SEP technology, i. e., 200 micron (0.008 in) thick cells and 150 micron (0.006 in) thick coverglass (Reference 6-1), as summarized in the second column of Table 6-2. The SEP characteristics are given in the first column, and the third column relates to deployable solar array systems that are discussed in Section 6.4. The differences in specific power of the blanket result from the differences in temperature for the noted output and in packing factors. The power platform array has a higher packing factor because it is a continuous roll.

The blanket weighs  $1.0 \text{ kg/m}^2$  ( $0.205 \text{ lb/ft}^2$ ), and the power is  $96.95 \text{ W/kg}$  ( $44.0 \text{ W/lb}$ ) at the operating temperature. These numbers are based on 200-micron (0.008 in) cells, because these cells represent the lowest weight (thinnest cells) achievable while avoiding the rapidly increasing cost of very

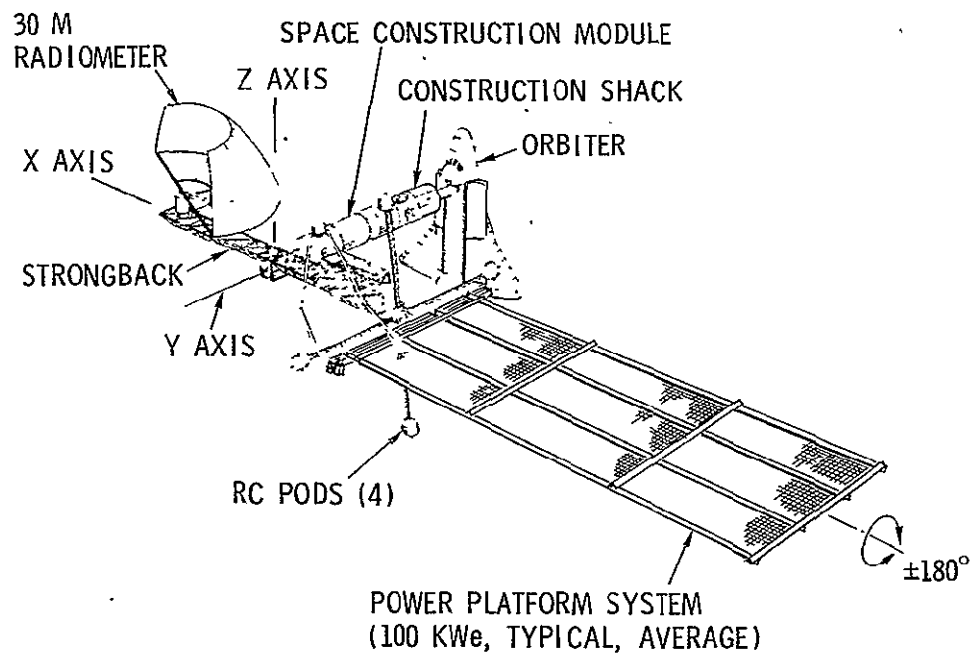


Figure 6-1. SCB Configuration, Fabrication and Assembly Solar Array

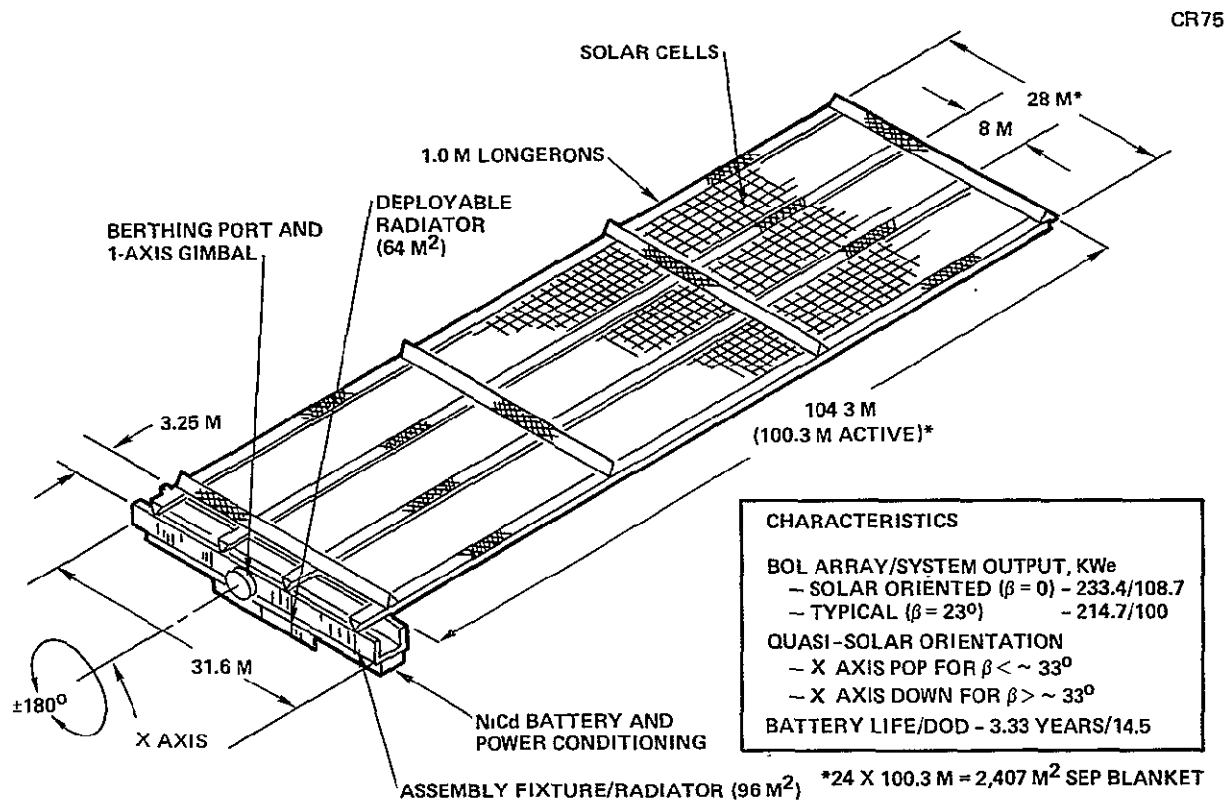


Figure 6-2. SCB Power Platform, Fabrication and Assembly Array

Table 6-2  
SOLAR COLLECTOR PERFORMANCE CHARACTERISTICS —  
SEP TECHNOLOGY<sup>(1)</sup>

|                                                                  | SEP<br>(1 Wing) | Power Platform | Deployable Solar<br>Array |
|------------------------------------------------------------------|-----------------|----------------|---------------------------|
| Array Output Power <sup>(2)</sup> , We                           | 12,500          | 233,400        | 230,700                   |
| Blanket Area, m <sup>2</sup> (ft <sup>2</sup> )                  | 125 (1345)      | 2407 (25,900)  | 2599 (27,965)             |
| Blanket Specific Power,<br>W/m <sup>2</sup> (W/ft <sup>2</sup> ) | 100 (9.29)      | 96.95 (9.010)  | 88.77 (8.25)              |
| Orbital Regime                                                   | GEO             | LEO            | LEO                       |
| Blanket Temperature, °C                                          | 55              | ~80            | ~80                       |
| Packing Factor                                                   | 0.812           | 0.887          | 0.812                     |

(1) Illumination of 1,353 W/m<sup>2</sup>

(2) BOL; conditions specified; 11.4% cells (AMO at 28°C); harness output

thin cells, as shown in Figure 6-3 (Reference 6-1). The SEP cell covers — 150 microns (0.006 in) thick — were selected as a reasonable compromise between weight and the increasing cost of thin covers, per Reference 6-1. The technology of thin plastic covers, e.g., 25 to 50 microns (0.001 to 0.002 in) of FEP Teflon or Spraylon, is considered inadequate for the SCB application; the SCB should use relatively conservative technology, as it is not an experiment but rather an important and expensive multipurpose support facility. Reference 6-1 (pp 6-7 and 6-8) also indicates the potential for cost savings of up to a few percent by going to (1) thicker cells and covers than the 200 micron/150 micron combination and (2) more efficient cells (based on current technology, fabrication approaches, and production demands). The slight savings do not appear sufficient to justify the production/availability risks associated with the more efficient cells, although this area warrants further study. In summary, the baseline design uses current SEP technology (200 micron-thick, wraparound cells, 150 micron-thick covers, and 11.4 percent solar cell efficiency (covered; 11.1 percent bare at 28°C).

Concentrating silicon photovoltaic systems were not considered because this unproven system has a higher risk related to long-duration mirror performance/degradation, higher blanket temperatures, nonuniform illumination,



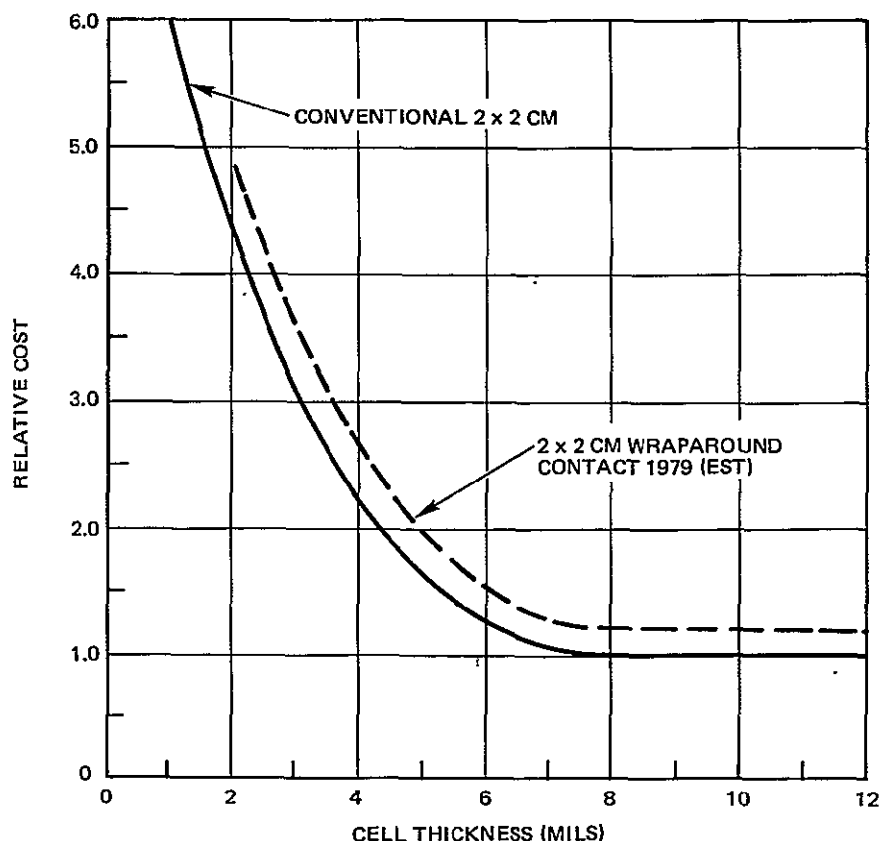


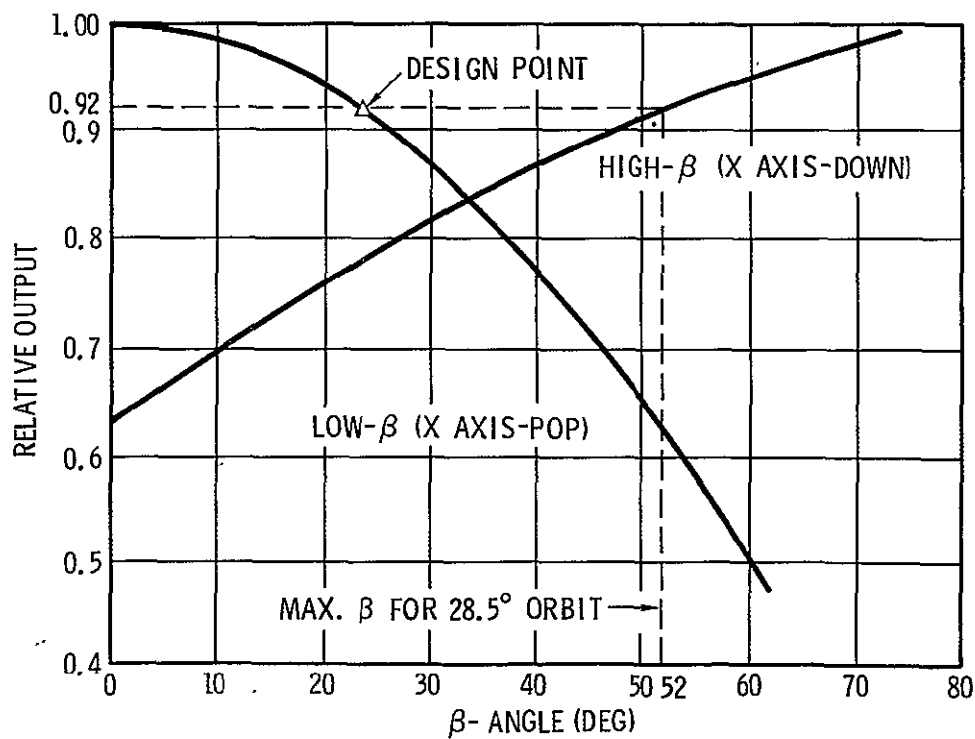
Figure 6-3. Thickness Impact on Solar Cell Cost

and more stringent pointing requirements. GaAs systems were not considered because of development status/risk and alignment/pointing accuracy problems associated with a large, relatively limber solar array. Some interesting options involving low-cost 200-micron-thick silicon cells are discussed in Section 6.4.3.

The solar array is sized for BOL because the prime early requirement for a capability of approximately 100-kWe average power is for TA-2 antenna testing, which will be completed early in the 10-year period. The other power requirements expected in the late 1980's (e.g., SPDF and MDL) do not fully utilize the 100-kWe capability.

Battery life is conservatively selected to be 3.33 years, based on 14.5 percent depth of discharge (DOD). The rationale for this selection is discussed in Section 6.3.3.

The array and system output, a function of the season of the year and SCB orientation, was summarized on Figure 6-2. The relative solar array output is shown in Figure 6-4 for two different SCB/solar array orientations, as noted. At  $\beta = 0$  degrees, the array axis (SCB X-axis) is perpendicular to the orbit plane (POP) and gimbals at orbit rate to keep the array always normal to the solar vector. This mode of array orientation is maintained as  $\beta$  increases, up to approximately 33 degrees; the array output drops off as shown. As  $\beta$  exceeds approximately 33 degrees, the SCB/solar array is reoriented so that the array axis (X-axis) is pointed at the center of the earth with the array in the orbit plane. The output increases as  $\beta$  increases to 52 degrees, the maximum value for a 28.5-degree-inclination orbit. The value of  $\beta$  cycles back and forth in the 0- to 52-degree region. A typical or average array output for these cycles is represented by  $\beta = 23$  degrees and a relative output of 0.92, the design point at which the array output is 214.7 kWe and the system output is 100 kWe. The system is rated at this point. The array output is 233.4 kWe at  $\beta = 0$  degrees, and the system output is 108.7 kWe, BOL. The SCB orientations described above are compatible with the SCB



$\beta$  = ANGLE BETWEEN SOLAR VECTOR AND ORBIT PLANE

Figure 6-4. Solar Orientation Effect on Power Platform Output

orientation requirements defined to date, which have not included the need for inertial orientations.

## 6.2 BASELINE POWER PLATFORM SYSTEM OPERATIONS

The solar array configuration defined in Figure 6-2 uses beams that are fabricated, essentially in place, of graphite-epoxy or graphite-polyimide composite materials from a locatable beam-fabrication module. The two-piece pallet for transporting the system and materials within the Orbiter is unfolded and mounted on a Construction Shack berthing port (or the Orbiter docking module in a sortie mode), where it becomes the fabrication and assembly fixture for the array (Figure 6-5, upper view). The array is built up entirely with the use of this single assembly fixture, which performs as the carrier pallet during boost phase for the system elements, and later becomes the primary structural interface and radiator for the power platform. It consists of two hinged (180-degree) channel-shaped structures. It is shown stowed in the Orbiter, together with the beam fabrication module, at the bottom of Figure 6-5. The mechanisms for holding and moving the longeron beams, the fittings for mounting the beam fabrication module, and the fittings for mounting the solar cell blanket rolls are contained within the channel structures.

The beam-fabrication module is removed from the launch location and placed at a beam position on the side, and the solar blanket rolls are relocated (Figure 6-6, left-hand sketch). The fabrication module is moved to each longeron position until all four are completed. The fabrication module is then located on the end of the fixture, and a transverse beam is fabricated and attached to the longerons. The solar cell blankets are attached to the cross beam, and the array is moved through the holding fixture. As the blankets unroll, they are periodically attached to the longeron via a system of negator springs. At an appropriate span, the array is stopped and another cross beam is fabricated and attached to the longerons (Figure 6-6, right-hand sketch). An overview of the SCB at this stage of construction is depicted in Figure 6-7. The SCB is powered by the 38-kWe power module. When the array is complete and the blankets are attached to the closeout cross beam, the longerons are rigidly attached to the pallet/assembly fixture.

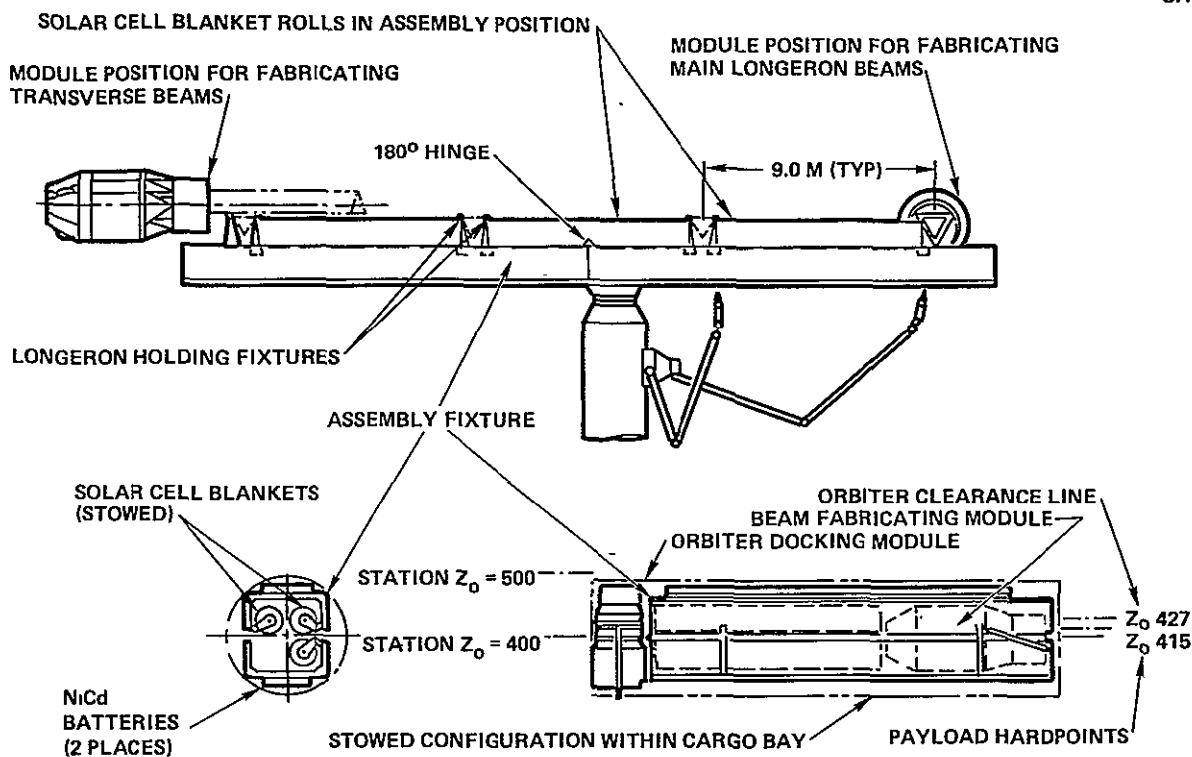


Figure 6-5. Fabrication and Assembly Tooling for 233-kW Power Platform

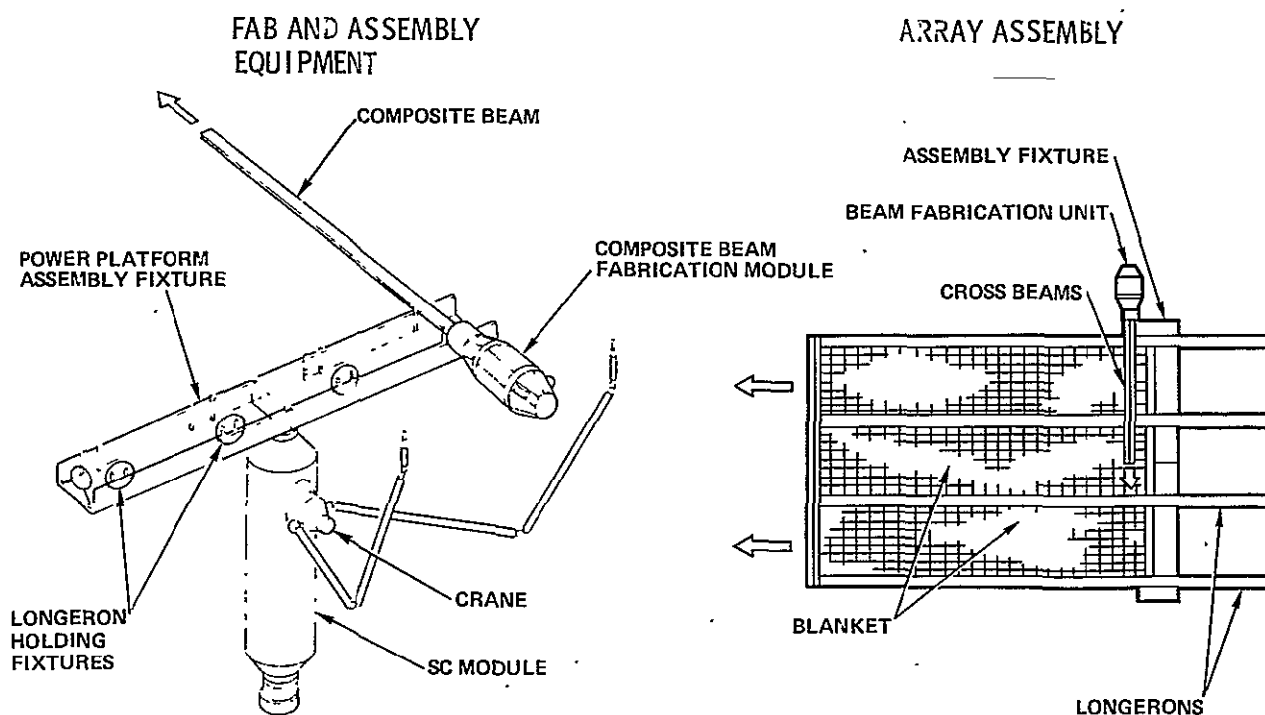


Figure 6-6. Solar Array Fabrication and Assembly

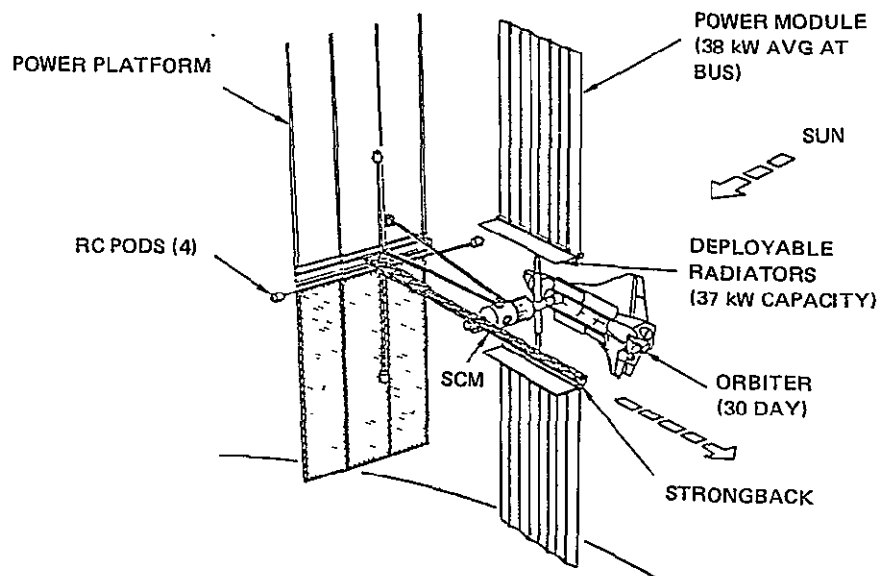
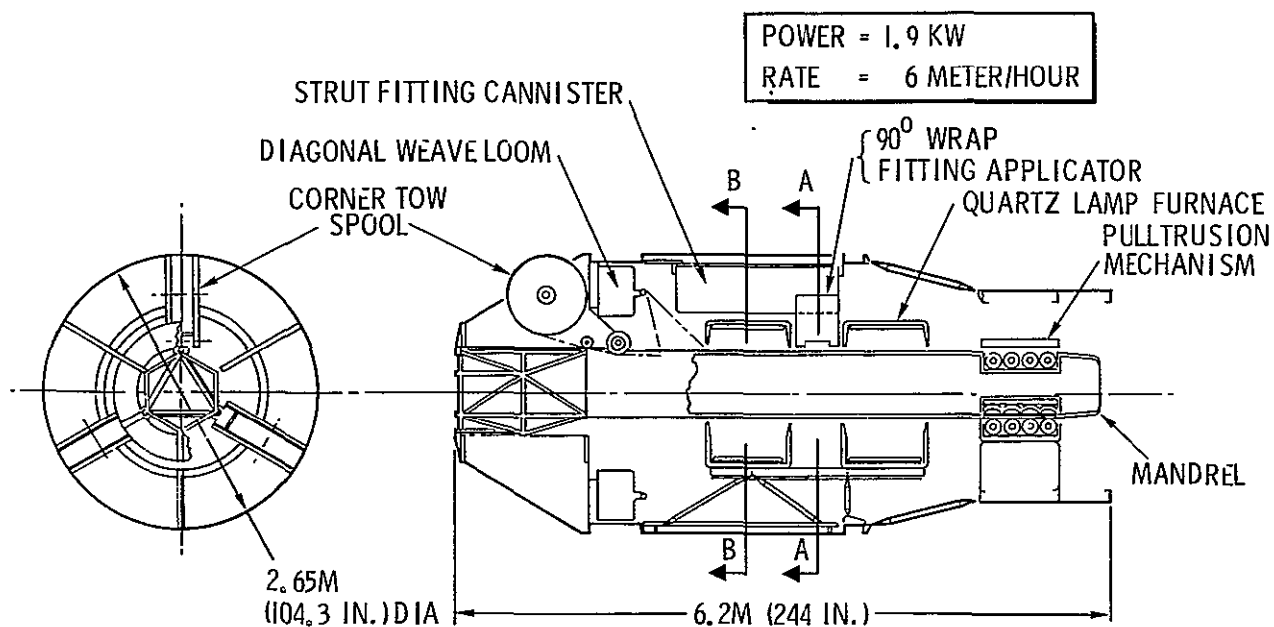


Figure 6-7. Power Platform Construction (Power Module-Supported)

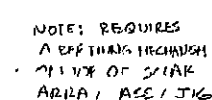
The beam fabrication module, which fabricates the 1-m triangular beams, is shown in Figure 6-8. It takes the preimpregnated composite material and wraps and/or weaves the material around the mandrel, while curing the beams on a continuous basis, until the desired length of longeron/cross beam is produced. Further details of the power platform construction and the beam fabrication module are presented in the Space Station Systems Analysis Study report, Part 3 (Reference 6-2, pp 4-15 to 4-20).

Because construction in an Orbiter sortie mode is considered as an alternative for the power systems considered in the study, an analysis was performed of how the baseline fabrication and assembly system might be constructed out of the Orbiter to provide a point of reference. The construction sequence (noted in Figure 6-9) is similar to that associated with using the SCB, as described above. An assembly fixture is delivered to orbit in a folded configuration. Stowed within this fixture are a beam fabrication module and solar array blanket rolls. The assembly fixture is installed on the Orbiter's docking module and then unfolded (deployed). The

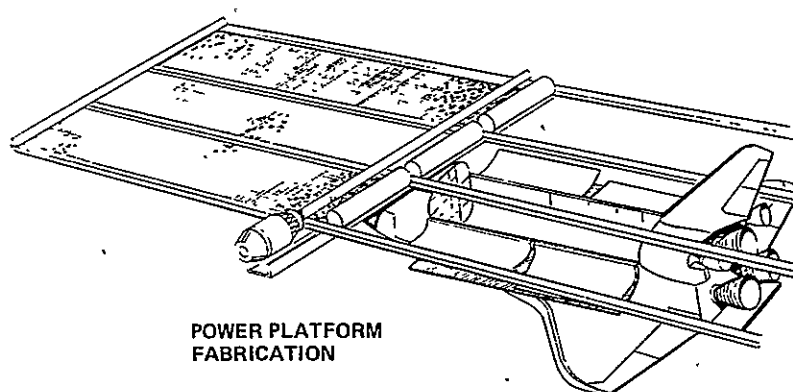


**Figure 6-8. Fabrication Module, 1-Meter Beam Composite Construction**

beam fabrication module is then maneuvered by the RMS to the end of the assembly fixture, where it is attached by an astronaut in extravehicular activity (EVA). The module is checked out, test coupons built, and the first 107.6-m longeron fabricated. The fabrication module is moved from position to position, and the second, third, and fourth longerons are fabricated. The RMS cannot reach the fourth position, so the module has to be positioned by either a second RMS installed in the starboard RMS location or by a special EVA handling fixture. The four longerons, having been "extruded" to a position forward of the Orbiter, are all run aft through the longeron guides. At the same time, main power buses are attached. The first cross beam is then fabricated and attached to the ends of the longerons. The three solar array blanket rolls are installed next, and the leader end of each is attached by means of tensioning fittings to the cross beam. The longerons are then moved forward through the longeron guides, and the solar blankets are allowed to unroll. The unrolling is stopped periodically to allow the solar array outputs to be connected to the power bus cables. This process is interrupted to allow fabrication and installation of a second cross beam as a support across the center of the array structure (Figure 6-10). The structure



**Figure 6-9. 233-kW Solar Array Fabrication and Assembly, Sortie Mode**



**Figure 6-10. Sortie Mode Configuration**

is closed out by fabrication of a third cross beam and attachment of the ends of the blankets.

At this point, a berthing port is installed on the top of the assembly fixture to allow for subsequent SCB buildup. A control system, which will stabilize the solar array adequately so that the next Orbiter can dock to it, is then activated and the Orbiter is separated. This process requires about 19 work days, assuming 2-shift operation, and thus requires a 3-week Shuttle capability. With the addition of contingency time and checkout requirements, this is increased to more than 30 days.

The next launch brings up the SCB and a docking adapter needed for the third launch, by which the strongback and solar array gimbal system are brought up so the final configuration may be assembled.

In analyzing this construction approach, it was found that the time needed to fabricate the solar array out of the Shuttle is about the same that would be



required to do it with the SCB. However, additional capabilities, over and above what are needed if the solar array is constructed using the SCB, are required, as summarized below:

- A second (berthing) port on the assembly fixture
- Attitude control incorporated in the assembly fixture
- A second RMS or special handling fixtures
- A docking adaptor
- Shuttle electrical power augmentation

### 6.3 ENERGY STORAGE SYSTEM EVALUATION

Several candidate energy storage system options were evaluated during Task 10 in conjunction with the baseline power platform solar array system shown previously in Figure 6-2. The energy storage options evaluated included (1) current-technology NiCd batteries, (2) advanced-technology NiCd batteries, (3) NiH<sub>2</sub> batteries, (4) regenerative fuel-cell/water-electrolysis (RFC), and (5) energy wheels (flywheels).

Energy wheels, which are discussed in Section 6.3.6, were evaluated together with the other systems before the midterm briefing, at which time they were dropped. Evaluation of the energy wheels showed the following: (1) poor development status and high development risk; (2) no significant weight advantage compared with the advanced NiCd and NiH<sub>2</sub> batteries, and a disadvantage when compared with regenerative fuel cells (RFC) despite operation at a high DOD (49 percent) which penalizes reserve capacity and load averaging capability, and lack of wheel rupture containment); (3) high expected DDT&E for development and long-life demonstration; (4) relatively short life expectancy (approximately ~1 to 2.5 years); at least early in the program; (5) relatively low energy storage efficiency, (6) potential SCB attitude control problems due to momentum imbalance between the two wheels of a counter-rotating pair; (7) self-discharge (rundown) in about 20 hours, which impairs emergency capability and long-term (e.g., 24-hour) load averaging; and (8) potentially high production and operations cost, because of the large number of relatively small units required.

### 6.3.1 Energy Storage System Summary and Comparison

Table 6-3 summarizes the characteristics of the remaining candidate systems. It should be noted that the table is for a system of 100-kWe average output and the baseline power platform solar array as was depicted in Figure 6-2. The array outputs (BOL) are as described in Section 6.1; the table values for the NiCd system are those in Figure 6-2. The differences in array output and array area result from the variations in storage efficiency. The RFC system has the poorest efficiency at 54.1 percent. However, the RFC array area is only 10 percent greater than for the advanced NiCd case, despite a 21-percent difference in energy storage efficiency. The efficiencies and array areas are derived for each of the candidates in Sections 6.3.2 to 6.3.5, which also discuss DOD, energy density, and expected life. The 33-percent

Table 6-3  
ENERGY STORAGE CHARACTERISTICS SUMMARY\*

|                                                              | NiCd       | Advanced<br>NiCd | NiH <sub>2</sub> | Regenerative<br>Fuel Cells |
|--------------------------------------------------------------|------------|------------------|------------------|----------------------------|
| Array Output, BOL, kWe                                       |            |                  |                  |                            |
| • Typical                                                    | 214.7      | 209.0            | 217.1            | 230.7                      |
| • Solar-Oriented                                             | 233.4      | 227.2            | 236.0            | 250.8                      |
| Array Area, m <sup>2</sup>                                   | 2,407      | 2,343            | 2,434            | 2,587                      |
| Storage Efficiency, %                                        | 62         | 65.7             | 60.8             | 54.1                       |
| Depth of Discharge, %                                        | 14.5       | 14.5             | 18.6             | 33                         |
| Energy Density <sup>(2)</sup> , W-h/kg                       | 3.93/27.08 | 6.39/44.1        | 9.49/51          | 25.0/75.1                  |
| Expected Life, Years                                         |            |                  |                  |                            |
| • Demonstrated                                               | 3.33       | None             | ~1               | 5+/3+ <sup>(1)</sup>       |
| • Design                                                     | 3.33       | 3.33             | 3.33             | 5                          |
| • Potential                                                  | 5-10       | 5-10             | 10               | 10                         |
| Peak Load Capability                                         | ~10X       | ~10X             | 2-10X            | ~4X                        |
| Load Averaging Potential                                     | FAIR       | FAIR             | FAIR             | GOOD                       |
| Launch Weight, kg                                            | 34,763     | 25,868           | 21,450           | 16,083                     |
| Resupply Weight (10 yr), kg                                  | 41,919     | 25,746           | 17,356           | 2,994                      |
| *100 kWe average at inverter output; baseline power platform |            |                  |                  |                            |
| <sup>(1)</sup> Fuel cell/electrolysis cell                   |            |                  |                  |                            |
| <sup>(2)</sup> Battery; usable/absolute                      |            |                  |                  |                            |

DOD for the RFC system relates to how much of the available gas storage capacity is used for each eclipse. The energy density is for a complete battery, not just cells; both usable energy density (the amount withdrawn at the operating DOD) and the absolute energy density (at 100 percent DOD) are shown. The demonstrated life for fuel cells is 5+ years, and 3+ years for the companion electrolysis cells. The design life is the value used for calculating mission replacement hardware requirements and the associated transportation costs. The potential life is an estimate of the life potential of the various systems. The regenerative fuel cells have good potential for 24-hour load averaging because the weight penalty for adding more gas tanks for storage is small. The launch weight is total weight launched to orbit for the initial installation; a breakdown of this weight is presented in Table 6-4. The resupply weight of Table 6-3 is the total energy storage hardware weight that must be resupplied during a 10-year mission. It is based on the replacement interval (design life), the weight of a replacement ship-set (as defined in Table 6-4), and an allowance for system spares.

The weight of Table 6-4 (and also Table 6-12 of Section 6.4) represents initial launch weight; it was derived using four sources of data. The first source was parametrics and detail values scaled from the MDAC Phase B Modular Space Station Study, when applicable (e.g., radiators, berthing ports, and gimbals). The second source was historical trend data from past and current programs (e.g., support structures, arrays, and thermal coatings). The third source was actual hardware, either as is or with modifications (e.g., NiCd batteries). The fourth source was those elements that are unique to this application. These include the large structural beam longerons/beams, which were based on structural design analysis.

The solar array (see Figure 6-2) consists of (1) the solar cell blanket, which assumes  $1.0 \text{ kg/m}^2$  ( $0.205 \text{ lb/ft}^2$ ) active area plus an allowance for the blanket portions blocked by the cross members, and (2) the array structure, which includes the four triangular composite longerons, four cross beams, attachments, and bus wire. The energy storage is, for example, the battery and its cold plate, but excluding the radiator and radiator loop. The structure/mechanical/miscellaneous includes (1) the assembly fixture/radiator ( $96 \text{ m}^2$ ), (2) the berthing port and 1-axis gimbal, and (3) the deployable radiator, the radiator fluid loops, and miscellaneous attachments. The

Table 6-4

POWER PLATFORM WEIGHT (FABRICATION AND ASSEMBLY ARRAY)

| Item                                   | NiCd                  | Advanced<br>NiCd | NiH <sub>2</sub> | Regenerative<br>Fuel Cell |
|----------------------------------------|-----------------------|------------------|------------------|---------------------------|
| Solar Array                            | 4,328                 | 4,199            | 4,367            | 4,653                     |
| • Blanket                              | (2,431)               | (2,367)          | (2,458)          | (2,611)                   |
| • Structure                            | (1,897)               | (1,832)          | (1,909)          | (2,042)                   |
| Energy Storage                         | 17,470                | 10,730           | 7,233            | 2,748                     |
| Structure/Mechanical/Miscellaneous     | 1,707                 | 1,505            | 1,400            | 1,450                     |
| Power Conditioning and Miscellaneous   | 2,014                 | 1,969            | 1,869            | 1,724                     |
| Composite Tube Weaving Unit            | 875                   | 875              | 875              | 875                       |
| Subtotal                               | 26,394                | 19,278           | 15,744           | 11,450                    |
| Contingency (25%)                      | 6,599                 | 4,820            | 3,936            | 2,863                     |
| EPS Total                              | 32,993                | 24,098           | 19,680           | 14,313                    |
| Orbiter Docking Module                 | 1,770                 | 1,770            | 1,770            | 1,770                     |
| Total, kg                              | 34,763 <sup>(1)</sup> | 25,868           | 21,450           | 16,083                    |
| Energy Storage Resupply <sup>(2)</sup> | 17,470                | 10,730           | 7,233            | 2,087                     |

<sup>(1)</sup>8,735 kg (50%) of batteries are off-loaded at launch

<sup>(2)</sup>One ship-set

power conditioning and miscellaneous item includes inverters, battery chargers, voltage regulators, wire, and miscellaneous switch gear and attachments. The beam fabrication module is shown in Figure 6-8.

A 25-percent contingency has been used uniformly in the SCB study, as it has been in this Alternate EPS Evaluation, Task 10. However, it should be noted that this imposes a somewhat unrealistic penalty on the solar array battery systems as compared to the reactor and solar brayton cycle systems, for which the technology is less mature. For example, the baseline NiCd battery system employs battery cells that have been fabricated and tested. Also, the SEP solar array blanket is a well-defined technology that already includes a contingency of 4 percent. The Orbiter docking module is very well defined and has been uniformly excluded from the contingency.

Because the NiCd battery system exceeds the Shuttle launch capability, 50 percent of the batteries (8735 kg) are brought up on a subsequent launch, at a cost penalty of approximately  $\$6 \times 10^6$ . A more realistic weight contingency and slight reoptimization of the battery DOD would likely allow launch of this system in a single Shuttle launch.

A summary of the total program cost attributable to the power system (for the fabrication and assembly power platform) is presented in Table 6-5 for the energy storage systems of interest. These costs include power system/vehicle integration costs. The costs for solar arrays were based on data obtained from Lockheed.

MDAC used information from its data bank to adjust the other costs required for the systems. This included structures, mechanisms such as gimbals and joints, environmental/thermal control equipment, electronics, power conditioning and distribution equipment, and attitude control subsystems. These costs were usually derived from the MDAC or Rockwell Phase B Space Station Studies and the recently completed Space Station Systems Analysis Study.

These cost estimates are based on general descriptions of the hardware and indicate the order of magnitude for the various items. More emphasis was

Table 6-5  
POWER PLATFORM ENERGY STORAGE OPTION COST\*

| Item                                | Cost (Million of \$) |               |         |                         |
|-------------------------------------|----------------------|---------------|---------|-------------------------|
|                                     | NiCd                 | Advanced NiCd | NiH2    | Regenerative Fuel Cells |
| DDT&E                               | (73)                 | (74)          | (75)    | (95)                    |
| Solar Array                         | 36                   | 36            | 36      | 36                      |
| Energy Storage/Power Conditioning   | 18                   | 19            | 20      | 40                      |
| Structure/Mechanical/Miscellaneous  | 19                   | 19            | 19      | 19                      |
| Production                          | (104)                | (104)         | (104)   | (96)                    |
| Solar Array                         | 70                   | 68            | 71      | 74                      |
| Energy Storage/Power Conditioning   | 32                   | 34            | 31      | 20                      |
| Structure/Mechanical/Miscellaneous  | 2                    | 2             | 2       | 2                       |
| Operations                          | (127.4)              | (114.5)       | (111.2) | (80.7)                  |
| Initial Launch Transportation       | 25.9 <sup>(1)</sup>  | 20            | 20      | 20                      |
| Replacement Hardware/Spares         | 33                   | 38            | 39      | 16                      |
| Replacement Hardware Transportation | 28.5                 | 17.5          | 11.8    | 2                       |
| Drag Propellant Transportation      | 40.0                 | 39.0          | 40.4    | 42.7                    |
| Subtotal                            | 304.4                | 292.5         | 290.2   | 271.7                   |
| Beam Fabrication Module             | 45                   | 45            | 45      | 45                      |
| Total                               | 349.4                | 337.5         | 335.2   | 316.7                   |

\*Fabrication and assembly array; power module supported

<sup>(1)</sup>One-half of batteries on subsequent launch.

placed on predicting the comparative costs of the various systems than on defining the absolute costs. In the implementing of this approach, uniform factors were applied across all items to convert from estimated manufacturing costs to total program cost.

The solar array item of Table 6-5 includes the solar cell blanket, the related support structure/attachments, and buses. The energy storage and power conditioning item includes the batteries, fuel cells, electrolysis units, chargers, regulators, inverters, switch gear, and wiring associated with distributing and controlling the power generated by the solar array and the energy storage system.

The structure/mechanical/miscellaneous item includes all hardware costs not included in the solar array and energy storage items, e.g., the assembly fixture/radiator, the berthing port, the 1-axis gimbal, the deployable radiator, and radiator fluid loops.

The operations costs include (1) initial launch costs for transportation to orbit of the initial ship set, at  $\$20 \times 10^6$  per Shuttle flight, (2) replacement hardware/spares procurement cost, (3) the cost of Shuttle transportation of replacement hardware and spares, and (4) an integration penalty for drag propellant transportation.

The spares and replacement hardware are calculated for a 10-year mission duration. It assumes the batteries have a 3-1/3-year life and the RFC system a 5-year life, as noted earlier. The spares are assumed to be 10 percent of the total cost on each of the three sets of batteries, 20 percent of the total cost of each of the two sets of fuel cells and electrolysis cells, and 25 percent of all other electrical items. (Note: this means the spares factor is 30 percent of the cost of a flight set of batteries, 40 percent of the cost of the fuel cells and 25 percent of the cost of the other items.)

The charts comparing various energy storage systems reflect the different costs associated with the development of the storage units and also of the power conditioning equipment required for them. The change in cost for the  $\text{NiH}_2$  batteries reflects a decrease in cost for developing the simpler charger

(offset by increased development cost for the batteries themselves). The development cost for the RFC system reflects an increase for the tankage and a decrease (by eliminating the battery chargers) as well as the change between the development cost of the NiCd batteries and the fuel cells.

The DDT&E difference of approximately  $\$20 \times 10^6$  for the RFC system and the battery systems is principally for the development of a new, large water electrolysis cell and a new, larger fuel cell (assumed here to be an adaptation of the Shuttle fuel cell, based on Shuttle cell technology, although the GE SPE technology is also a good candidate).

The use of minimum-modification Shuttle fuel cells offers a very low DDT&E cost, at the expense of considerably higher production and operations costs because of the large number of small units involved and the heavier weight to be transported. The DDT&E investment in larger fuel cells is more than repaid by the production and operations savings. An indication of this can be seen in Table 6-6 for the P&W Shuttle ( $\$5.1 \times 10^6 + \$20.7 \times 10^6 = \$25.8 \times 10^6$ ) and the modified PC-15 ( $\$9.5 \times 10^6 + \$8.3 \times 10^6 = \$17.8 \times 10^6$ ). These costs do not include the operational savings for the PC-15 option. The nonrecurring costs in Table 6-6 include the costs of system integration testing and the associated hardware requirements. The costs are adjusted as noted on the table based on inputs received (in response to MDAC SCB requirements) from Eagle-Picher, P&W, and GE. Water electrolysis cell DDT&E is a large cost item in Table 6-6.

The solar array DDT&E of Table 6-5 is largely analysis and design, as only limited development/system integration testing is practical. The structure/mechanical/miscellaneous DDT&E is also solar array related. The energy storage/power conditioning DDT&E includes development of all the power conditioning equipment and considerable system integration testing, as well as development of the energy storage device(s), per se.

The production cost of the solar array reflects the change in the solar array area associated with the various systems. The production energy storage cost covers the cost of the initial hardware (1 ship-set); the RFC system offers relatively low cost, because only a few large units are required as the result of the DDT&E investment noted previously.



Table 6-6  
ENERGY STORAGE IMPACT OF VENDOR COSTS ON PROGRAM\*

|                                                                                                 | Quantity In<br>One Ship-Set | Nonrecurring | Recurring<br>(One Ship-Set) |
|-------------------------------------------------------------------------------------------------|-----------------------------|--------------|-----------------------------|
| Batteries                                                                                       |                             |              |                             |
| • NiCd                                                                                          | 128                         | 8.1          | 12.1                        |
| • Advanced NiCd                                                                                 | 128                         | 9.4          | 14.2                        |
| • NiH <sub>2</sub>                                                                              | 128                         | 10.5         | 15.9                        |
| Fuel Cells                                                                                      |                             |              |                             |
| • P&W Shuttle                                                                                   | 32                          | 5.1          | 20.7                        |
| • P&W, Improved Shuttle                                                                         | 32                          | 11.3         | 19.1                        |
| • P&W, PC-15                                                                                    | 8                           | 9.5          | 8.3                         |
| • GE, 14.3 kWe                                                                                  | 8                           | 7.2          | 1.2                         |
| Water Electrolysis Cells                                                                        |                             |              |                             |
| • GE, 15 kWe                                                                                    | 8                           | 14.3         | 2.3                         |
| *Millions 1978 \$; vendor costs adjusted for quantity required, inflation, and program factors. |                             |              |                             |

The initial launch transportation cost covers transportation to orbit for the initial system. All systems fully utilize one Shuttle launch ( $\$20 \times 10^6$ ), except the NiCd system, which requires part of a second launch because its weight exceeds the Shuttle capability, as noted earlier. The replacement hardware/spares operational cost is based on the cost of a ship-set of hardware and on hardware life, which sets the number of replacement systems needed. The RFC system shows a big advantage here, because of long life and the relatively few units per ship-set. Replacement hardware transportation is the transportation cost for the replacement hardware, which depends on hardware life and weight per replacement ship-set. The RFC system is particularly advantageous here due to long life and low weight. Drag propellant transportation costs reflect variations in array area for the various systems.

This analysis is based on an assessment of system differences ( $\Delta$ 's), and the differences in totals are real, even though they may be on the same order as the accuracy of the absolute values of the totals. The RFC system shows a significant saving of  $\$18$  to  $32 \times 10^6$  over the other systems. Discussions with F. E. Ford (Reference 6-3) of NASA GSFC indicated that NiCd battery experience suggests lifetimes of 5 to 10 years are possible at 15-percent DOD with  $0^\circ\text{C}$  average battery temperatures. Subsequent attempts to obtain supporting data were unsuccessful; however, the following table summarizes the effect on total system costs (analogous to Table 6-5) if battery lifetimes of 3.33, 5, and 10 years were to be obtained.

|                                | BATTERY LIFE (YEARS) |       |       |               |       |       |
|--------------------------------|----------------------|-------|-------|---------------|-------|-------|
|                                | NiCd                 |       |       | Advanced NiCd |       |       |
|                                | 3.33                 | 5     | 10    | 3.33          | 5     | 10    |
| Total<br>Cost<br>(\$ Millions) | 349.4                | 323.7 | 300.0 | 337.5         | 314.7 | 293.9 |

The RFC system cost is approximately equal to that of the NiCd batteries at 5-year life. The 10-year life NiCds show a \$17 to \$20 million advantage over the RFC system. Estimation of the life of the various advanced tech-

nology options is difficult and requires further study. The operational costs generally constitute the largest category and should be carefully evaluated.

Sections 6.3.2 through 6.3.6 describe each of the candidate energy storage systems.

#### 6.3.2 Current Technology NiCd Batteries

This section describes the current technology NiCd battery system application. A block diagram of the system with a NiCd battery is presented in Figure 6-11. The 100-kWe system is comprised of eight identical electrical modules like the one shown, each of which provides 12.5 kWe of average load power at 400 cycles ac. A 400-cycle ac system was selected as a typical power conditioning and distribution approach based on the JSC/Rockwell International Phase B Space Station study results. A JSC guideline for the MDAC Space Construction Base (SCB) studies was to use the Rockwell design baseline as a point of departure unless it was clearly inappropriate. MDAC felt that the Task 10 resources could be best utilized to evaluate energy storage and solar array options, rather than to reevaluate the Rockwell power conditioning and distribution trades. The energy storage and solar array areas have greater SCM impact.

The energy storage portion of the system is within the dotted lines of Figure 6-11 and will be discussed subsequently. A dc/dc regulator is used as the energy storage device output in all cases, as in the Rockwell Phase B system. Figure 6-11 shows the energy storage system output power (13.44 kWe) and energy (8.06 kW-hr) during the 36-minute (0.600-hour) night period. The energy storage system input is 13.40 kWe and 12.95 kW-hr for the 58-minute (0.967-hour) worst case sunlight period. The solar array output is 26.84 kWe (BOL), which requires an area of 300.9 m<sup>2</sup> for this 1/8th of the total system. The array area is for the baseline power platform solar array concept. The  $8 \times 300.9 \text{ m}^2 = 2,407 \text{ m}^2$  array provides 100 kWe average to the system at  $\beta = 23$  degrees as discussed earlier.

The details of the energy storage subsystem (within the dotted lines) are presented on Figure 6-12, which shows the power, energy, current, voltage, and heat rejection at various locations in the subsystem. Again, the figure

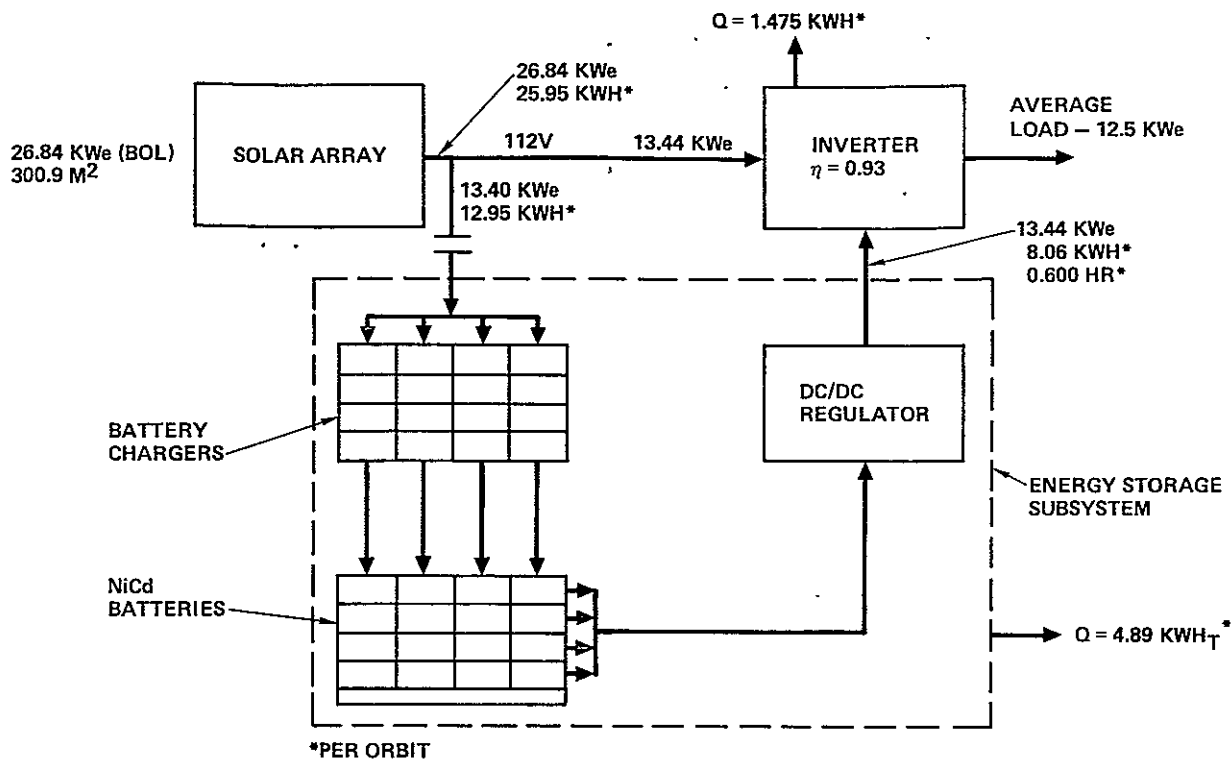


Figure 6-11. Solar Array NiCd Battery System (1 of 8, 12.5 kWe Each, Fab and Assy)

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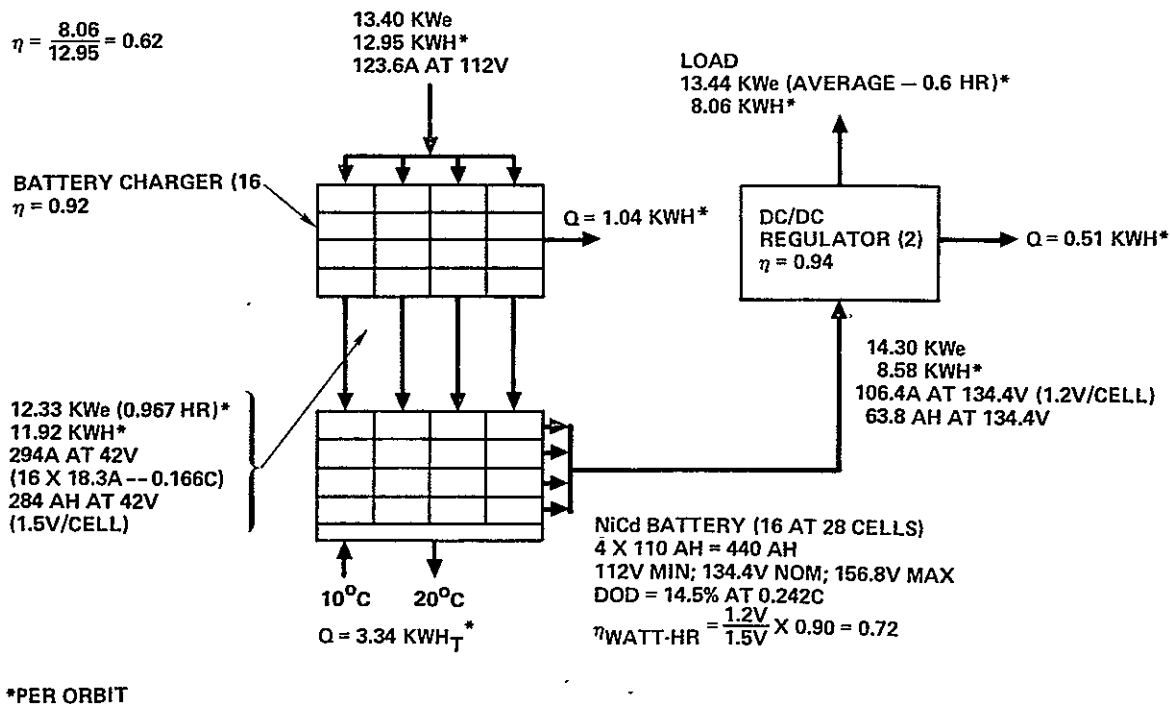


Figure 6-12. NiCd Battery Energy Storage (1 of 8)

represents 1/8th of the total 100-kWe system. The system has 16 batteries of 28 cells each, which allows spare cells for reliability reasons. Each battery is 110 A-hr capacity and four are discharged in series to obtain a nominal 134.4 volts (1.2 volts per cell) and a minimum 112 volts (1.0 volts per cell). The battery is the one developed by JSC during and subsequent to the Phase B Space Station study. The nominal DOD is 14.5 percent for life reasons, to be discussed below; the typical drain rate is 0.242 C. The energy efficiency of 72 percent is based on an A-hr efficiency of 90 percent and 1.2 volts average discharge voltage and 1.5 volts average charge voltage. These voltages were selected conservatively to allow for some degradation of voltage performance with age for the 3.33-year life. Hence, the system is designed with sufficient array capacity to meet all system loads and adequately charge an old and somewhat degraded battery.

The cooling requirement for these 16 batteries is 3.34 kW-hr/orbit. The heat is dissipated, largely during battery discharge, to a pumped loop cold plate, similar to the MDAC Skylab system. The battery cooling temperatures are consistent with the MDAC Phase B Space Station study results and those of the JSC 110 A-hr battery development, although it appears that the SCB orientations and radiator configurations presently envisioned permit lower battery temperatures (e.g., 0-10°C). Each battery has its own charger and charges at the 0.166 C° rate. The overall energy storage subsystem efficiency is 62 percent, as noted in the upper left corner of Figure 6-14, based on subsystem output and input energy.

### 6.3.3 Advanced Technology NiCd Batteries /NiCd Lifetime

Figure 6-13 is a block diagram of the advanced NiCd battery system and Figure 6-14 is a block diagram of the energy storage portion of this system. The primary difference between this system and the previously discussed NiCd battery system is a smaller solar array resulting from a higher energy storage system efficiency, 65.7 percent. The A-hr efficiency is assumed to be 95 percent for the advanced NiCd system, based on Reference 6-4.

The selection of battery system life and energy density is the subject of the ensuing discussion. A battery discharge curve for the JSC 110 A-hr cell, based on the MDAC test data, is presented in Figure 6-15. This curve is based on beginning-of-life (BOL) performance characteristics. At 15 percent

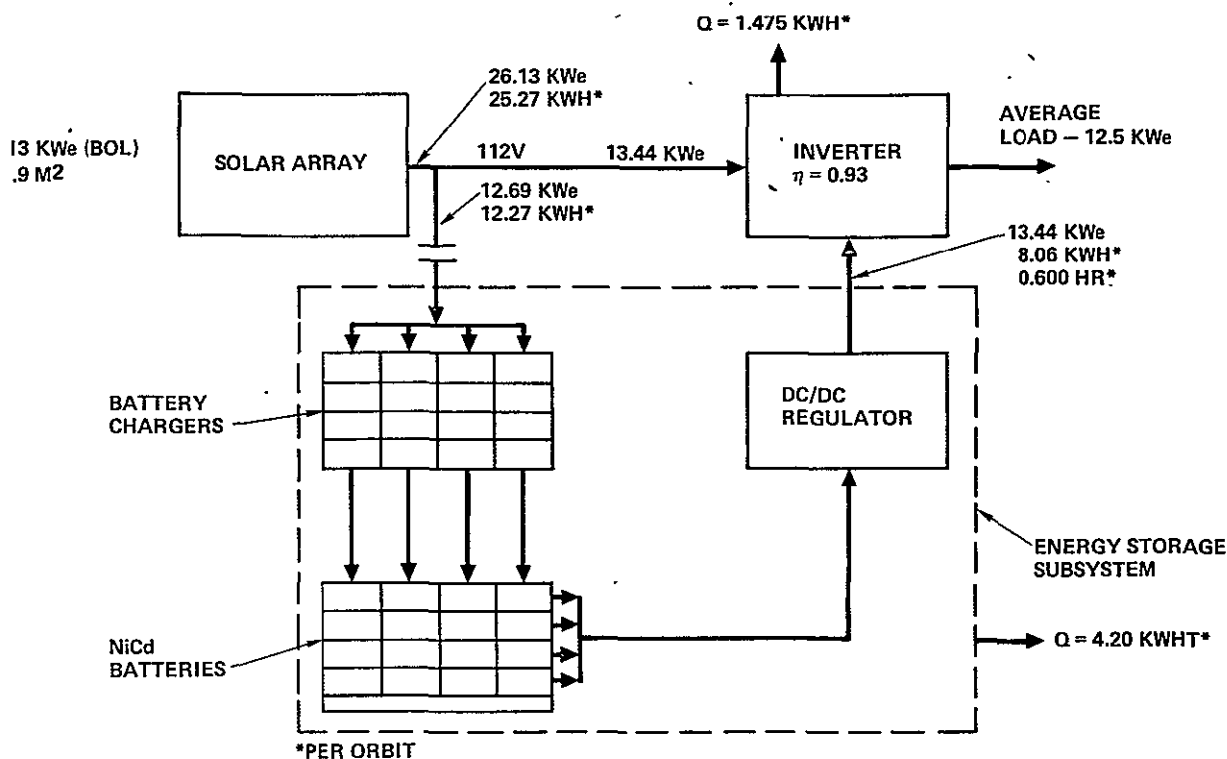


Figure 6-13. Solar Array Advanced NiCd Battery System (1 of 8, 12.5 kWe Each, Fab and Assy)

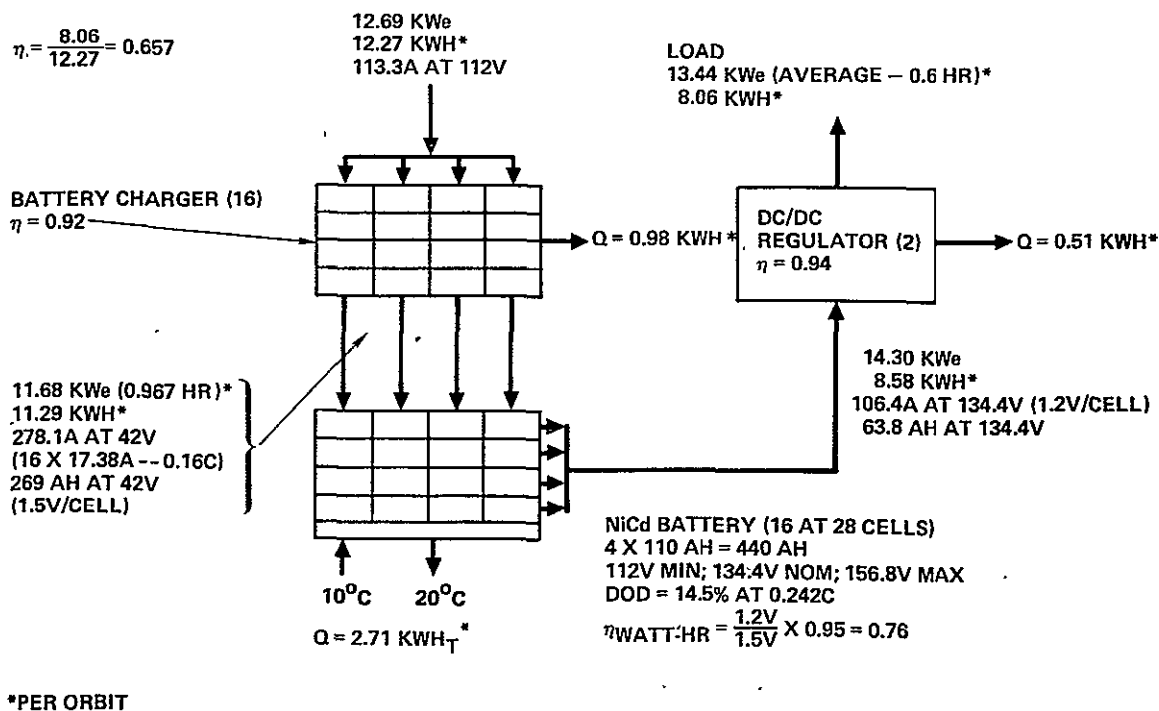


Figure 6-14. Advanced NiCd Battery Energy Storage (1 of 8)

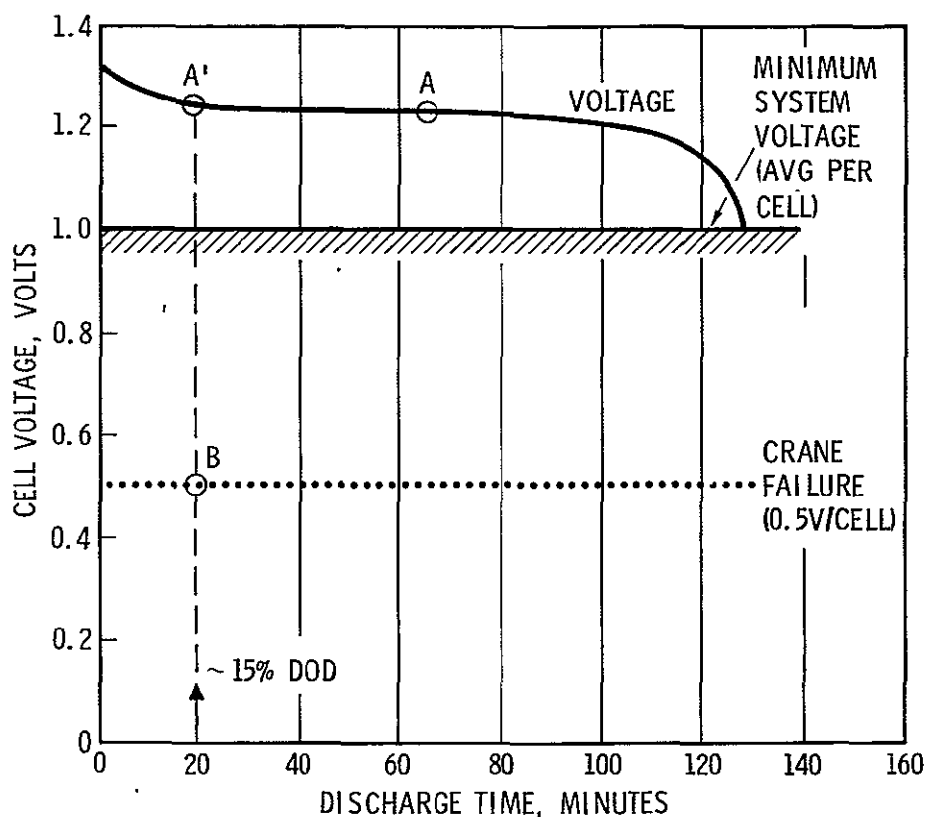


Figure 6-15. NiCd Battery System Utility

depth-of-discharge (DOD), the battery would typically be discharged to Point A'. In addition, the battery would frequently be discharged to Point A for either peak load situations or 24-hour load averaging. The system is designed to operate satisfactorily with 1.0 volt per cell, which provides a margin for (1) peak loads at higher drain rates, (2) energy requirements beyond Point A and (3) degradation of performance characteristics with age. The cell failure criteria utilized in the NWSC Crane test program is 0.5 volt per cell, as illustrated at Point B at 15 percent DOD. Degradation from Point A' to Point B is quite severe and clearly unacceptable if many cells experience this amount of degradation, because of the system design voltage of 1.0 volt per cell average. Consequently, it is not realistic to utilize NWSC cycle life data directly, as the 0.5 volt per cell is unacceptable for system design purposes. Hence, system design life should be considerably less than indicated by NWSC cycle life data.

NWSC defines failure of a battery pack (either 5 or 10 cells) as the point at which 60 percent of the cells in the pack have degraded to 0.5 volt per cell. MDAC has analyzed the distribution of cell failures prior to pack failure,

as indicated in Figure 6-16. For example, the curve shows that 60 percent of the cells have failed when the cycles to pack failure is 100 percent. Another point of interest on the curve is that at which 10 percent of the cells have failed, at which time the typical pack has experienced 59 percent of its cycles to failure. This is a reasonable design point for system design purposes. However, it means that the NWSC cycle life data must be reduced to 59 percent of the pack failure cycle life, for system life prediction purposes. The data plotted in Figure 6-16 assumes that all failures at less than the average life are excluded from the data.

Temperature and battery DOD are important parameters affecting cycle life. These parameters have been plotted in Figure 6-17 using the MDAC definition of battery cycle life as the point where 10 percent of the cells have a terminal voltage less than 0.5 volt. Figure 6-17 is also based on the best 50 percent of the NWSC data, as mentioned previously. The SCB design point is 15°C. A DOD of 15 percent results in a cycle life of 20,000 cycles, or approximately 3.5 years. Lowering the temperature to 0-5°C, would improve the life to approximately 4.3 years. Battery life intervals of 3.33, 5, or

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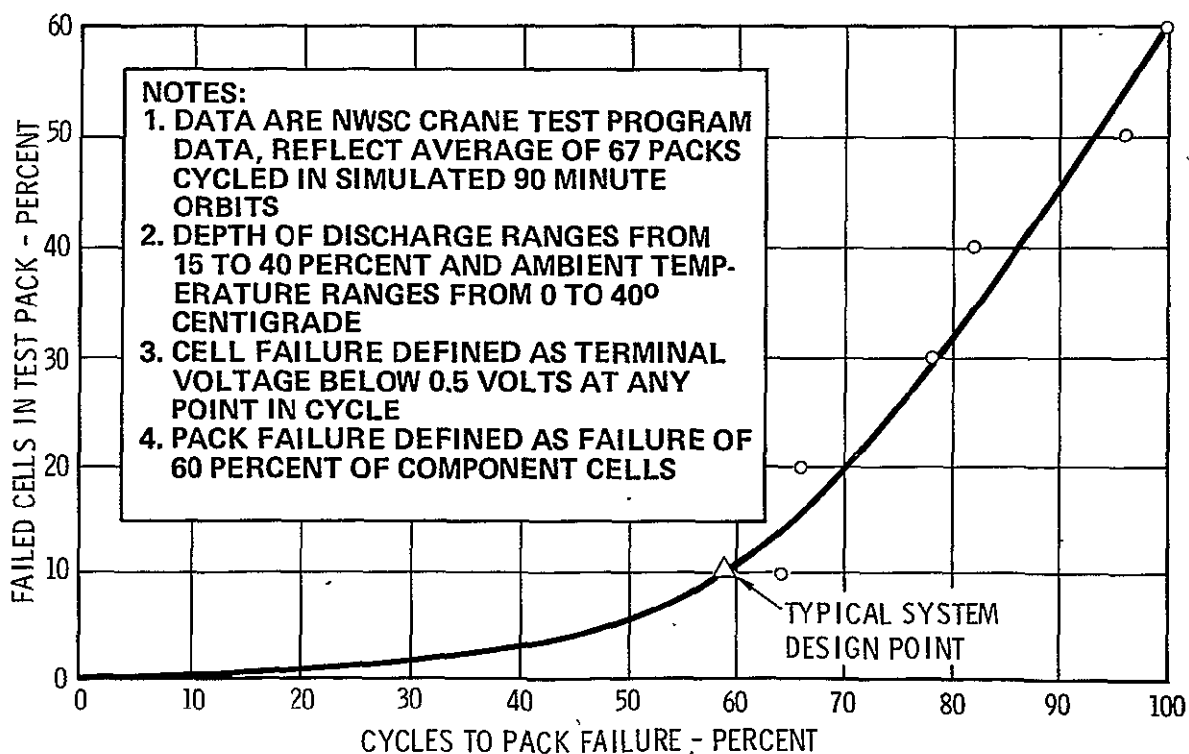


Figure 6-16. Average Cell Failure History vs Cycles to Pack Failure



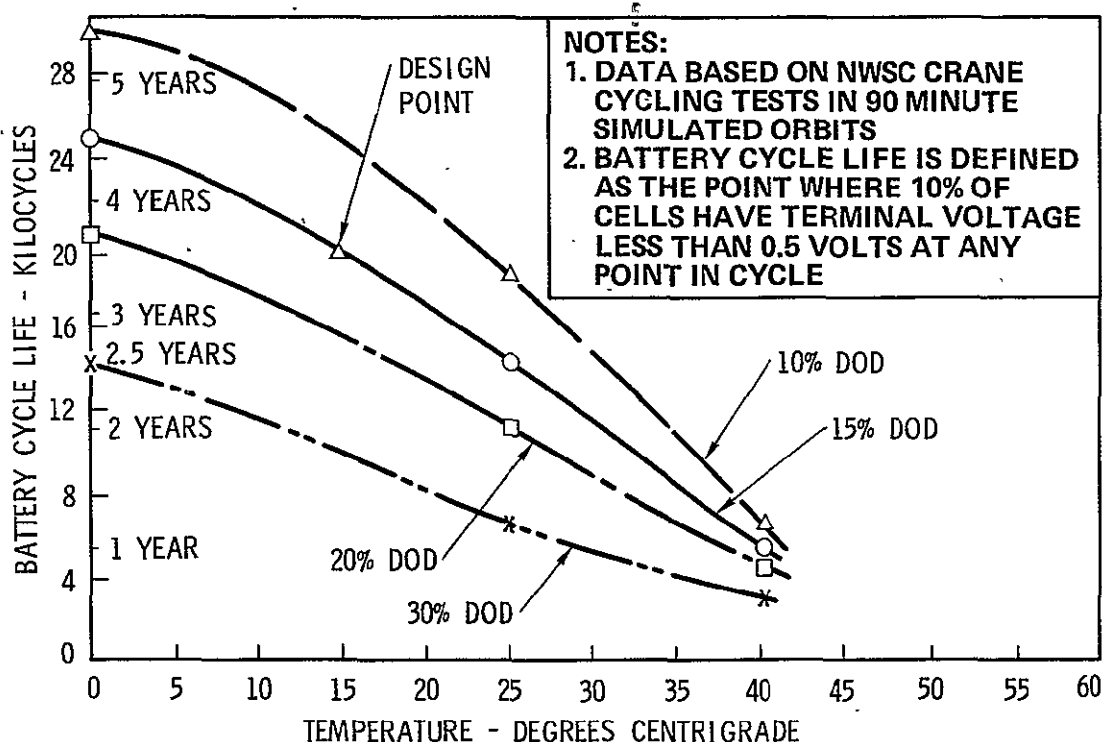


Figure 6-17. Battery Cycle Life as a Function of Temperature and Depth of Discharge

10 years are most appropriate for a 10-year mission. MDAC has selected a NiCd battery life of 3.33 years, based on Figure 6-17, and the desire for a degree of conservatism because of the NWSC failure criteria of 0.5 volt per cell. This is an area requiring further study.

A summary of the NiCd battery life and energy density characteristics for both present technology and advanced technology cells is presented in Figure 6-18. The MDAC SCB design point for both batteries and cells is shown at 3.3 years life and approximately 5 W-hr/kg energy density. The curve "MDAC - Present Cells" is based on the battery life analysis discussed in Figures 6-16 and 6-17 and the JSC 110 A-hr cell, which provides 36 W-hr/kg at 100 percent DOD. The curve labeled "MDAC - Advanced Cells" is based on the previously discussed MDAC life analysis and an energy density of 55 W-hr/kg at 100 percent DOD. The 55 W-hr/kg number is based on Reference 6-4, which usually refers to 55 W-hr/kg for advanced technology cells. MDAC feels that 44-49 W-hr/kg is a more realistic number for advanced technology cells that must operate in low earth orbit

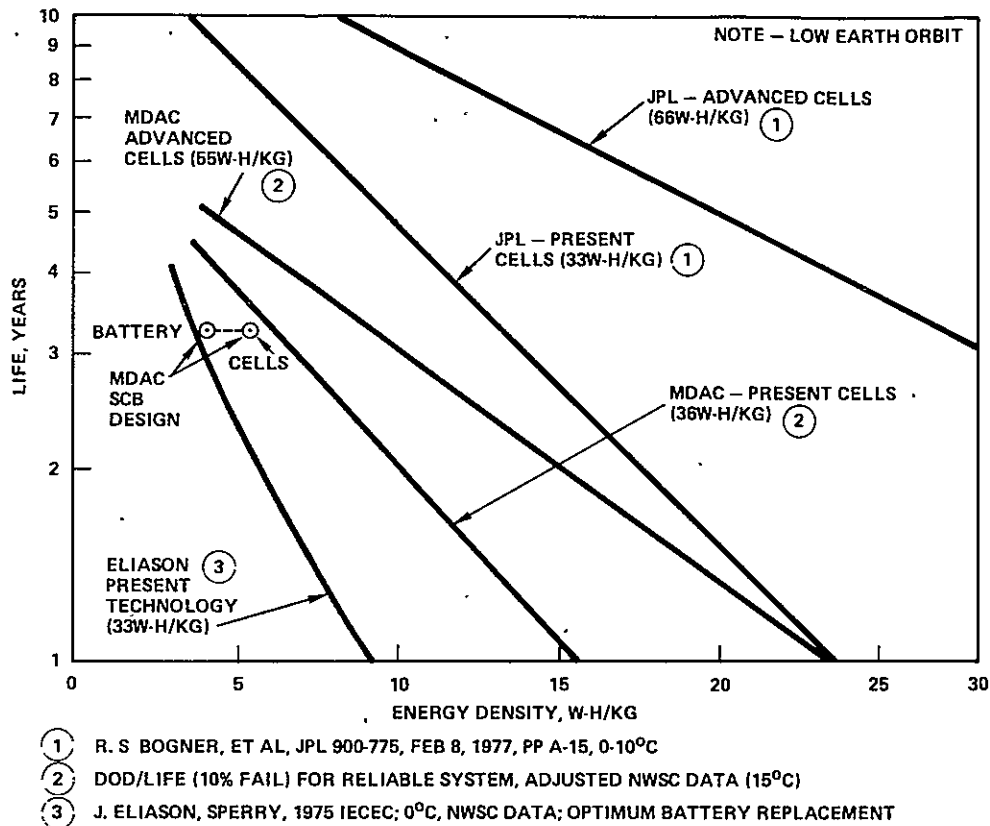


Figure 6-18. NiCd Battery Performance

(LEO) with relatively high drain rate requirements. The "JPL - Present Cells" and "JPL - Advanced Cells" curves are from References 6-4 and 6-5.

Reference 6-5 states that "the state-of-the-art cell life versus energy density curves are based on the NASA test program conducted at Crane NWSC, as summarized by Fono. The curves represent average cell life." It is further stated that the life and energy density objectives are "basically aimed at geosynchronous or planetary missions." The JPL curves assume an optimum temperature range of 0-10°C. Average cell life does not appear to be sufficiently conservative and consistent with SCB EPS system design requirements as discussed earlier in conjunction with Figures 6-15 through 6-17.

Eliason (Reference 6-6) presented data on the optimum replacement interval as a function of battery DOD. This interval was 2.3 years at 15 percent DOD for a 24-cell battery, assuming a cell energy density of 33 W-hr/kg at 100 percent DOD; this results in the curve labeled "Eliason Present Technology."

In summary, MDAC has selected a battery life of 3.33 years to provide a margin of safety with respect to NWSC test data and failure criteria. This is an area requiring further study. Battery temperature has been shown to be a key variable in life calculations. Battery and fuel cell thermal control radiators for the SCB are discussed in Section 6.3.7.

#### 6.3.4 NiH<sub>2</sub> Battery System

An overall EPS system block diagram for the NiH<sub>2</sub> system is presented in Figure 6-19. The energy storage subsystem diagram is shown in Figure 6-20. The NiH<sub>2</sub> battery energy storage system efficiency is 60.8 percent, which is similar to the NiCd battery efficiency. The cell size is assumed to be 100 A-hr and four batteries are discharged in series to obtain a relatively high input voltage for the dc/dc regulator. The DOD was selected to be 18.6 percent, a number close to that used for the NiCd batteries. The battery cooling temperatures were assumed to be the same as for the NiCd battery although the NiH<sub>2</sub> battery is more tolerant of elevated temperatures. Each battery has its own battery charger, which is simplified compared to the NiCd battery charger. The NiH<sub>2</sub> cell has excellent tolerance to overcharge and cell reversal.

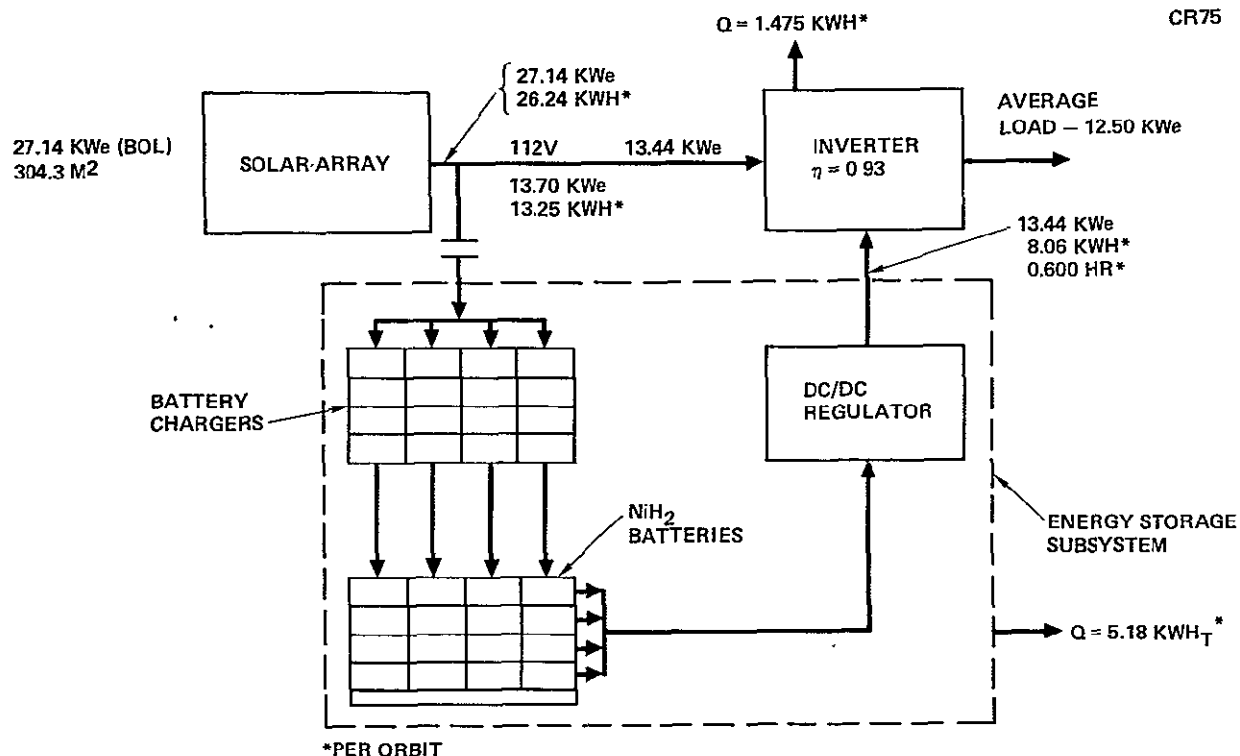
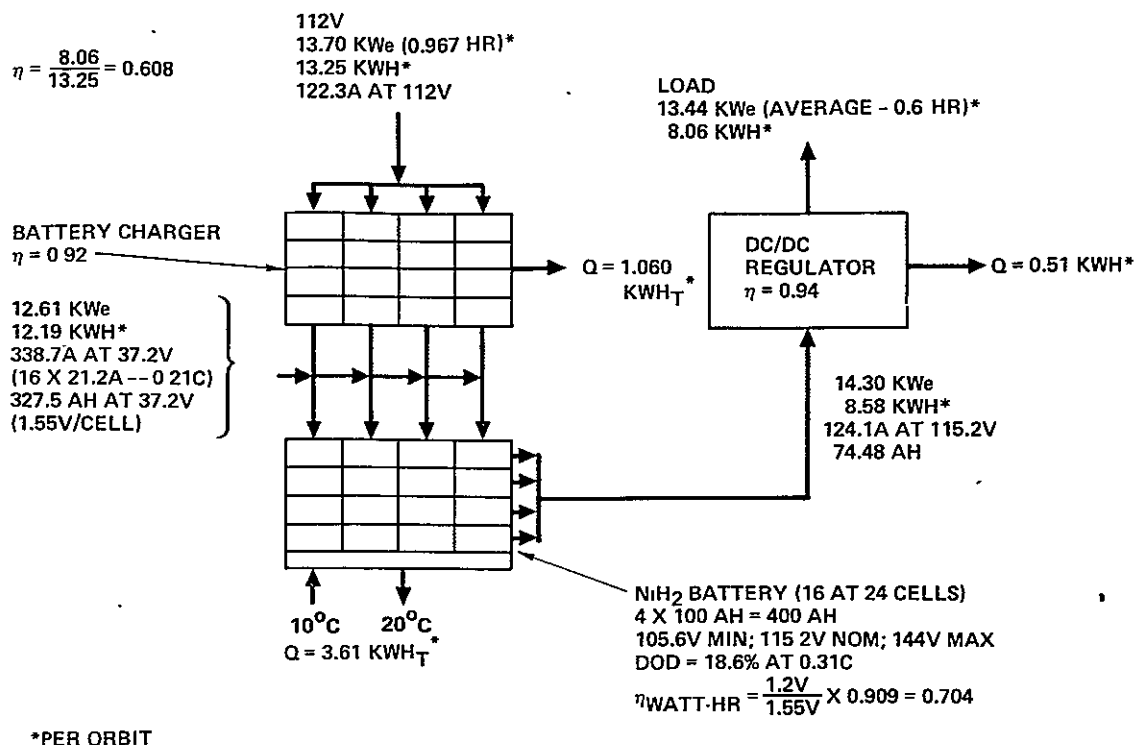


Figure 6-19. Solar Array NiH<sub>2</sub> Battery System (1 of 8, 12.5 kWe Each, Fab and Assy)

Figure 6-20. NiH<sub>2</sub> Battery Energy Storage (1 of 8)

There is very little life test data for the NiH<sub>2</sub> battery, particularly for LEO applications. This is understandable because the NiH<sub>2</sub> batteries are a new concept. Figure 6-21 is based on data presented by Rusta (Reference 6-7). The figure is basically for geosynchronous applications, but the one data point at 30 percent depth of discharge is for an accelerated LEO test (0.5 hour for the complete cycle). This data point indicates that a one-to-two-year life for NiH<sub>2</sub> has been demonstrated on a preliminary basis. However, the NiH<sub>2</sub> battery has potentially superior features with respect to NiCd and has excellent potential for long life, perhaps 5 or 10 years. A life of 3.33 years was selected for SCB design purposes as a compromise between the very limited life demonstrated to date and the excellent potential for long life. Timely life testing for LEO conditions is of extreme importance. There is also concern for electrolyte location stability in zero-g if non-wetting separators are used. It may be appropriate to conduct an orbital flight test, perhaps LDEF, to verify zero-g electrolyte-separator suitability.

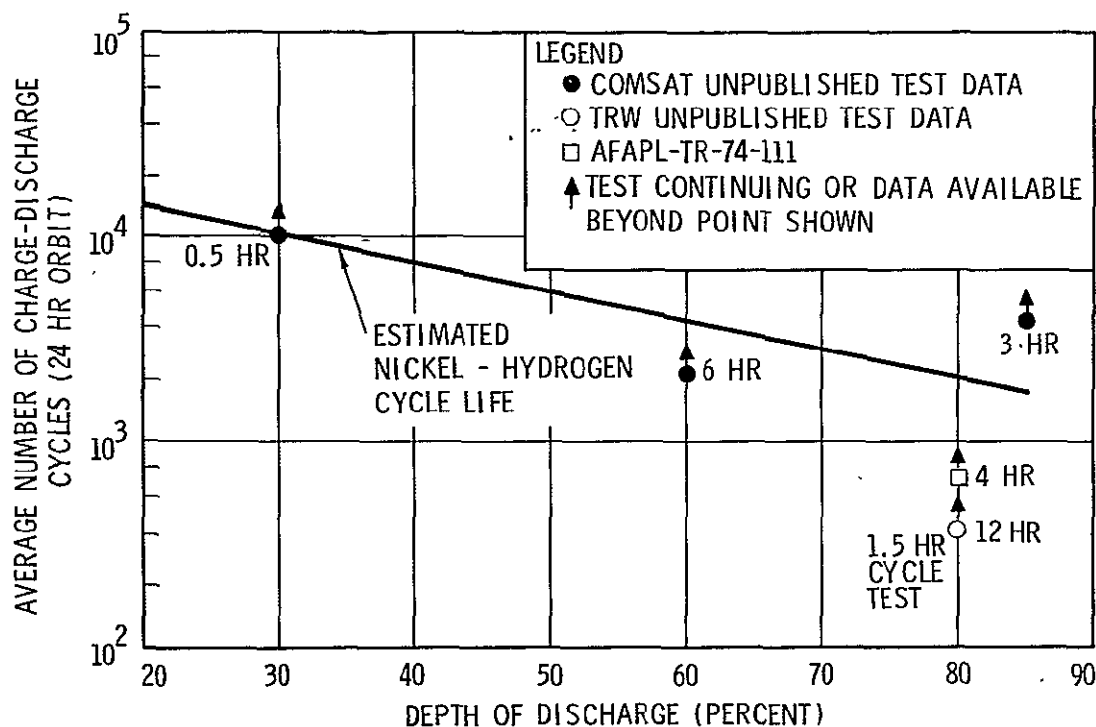


Figure 6-21.  $\text{NiH}_2$  Battery Life Trends

#### 6.3.5 Regenerative Fuel Cells

A block diagram of the regenerative fuel cell (RFC) system, which incorporates a fuel cell and a water electrolysis unit, is presented in Figure 6-22. The energy storage portion of the RFC system is presented in Figure 6-23. The solar array for this system is approximately 10 percent larger than for the battery systems because of lower energy storage system efficiency. The fuel cell consumes gaseous  $\text{H}_2$  and  $\text{O}_2$  during the dark portion of the orbit to provide load power. The fuel cell water is stored in a water tank for subsequent electrolysis during the sunlit portion of the orbit. The water electrolysis cell replenishes the  $\text{GH}_2$  and  $\text{GO}_2$  tanks. The system depicted in Figure 6-23 employs a single fuel cell and a single electrolysis cell: eight such systems are required to provide the average load requirement of 100 kW.

A single large cell is preferable to four smaller Shuttle-size cells for cost reasons, as discussed in Section 6.3.1. The proposed fuel cell operates at lower temperatures than the Shuttle fuel cell for life reasons as will be discussed below. The  $\text{GH}_2$  and  $\text{GO}_2$  tanks operate in the range of approximately 100-400 psi. These tanks are sized to accommodate a total inventory

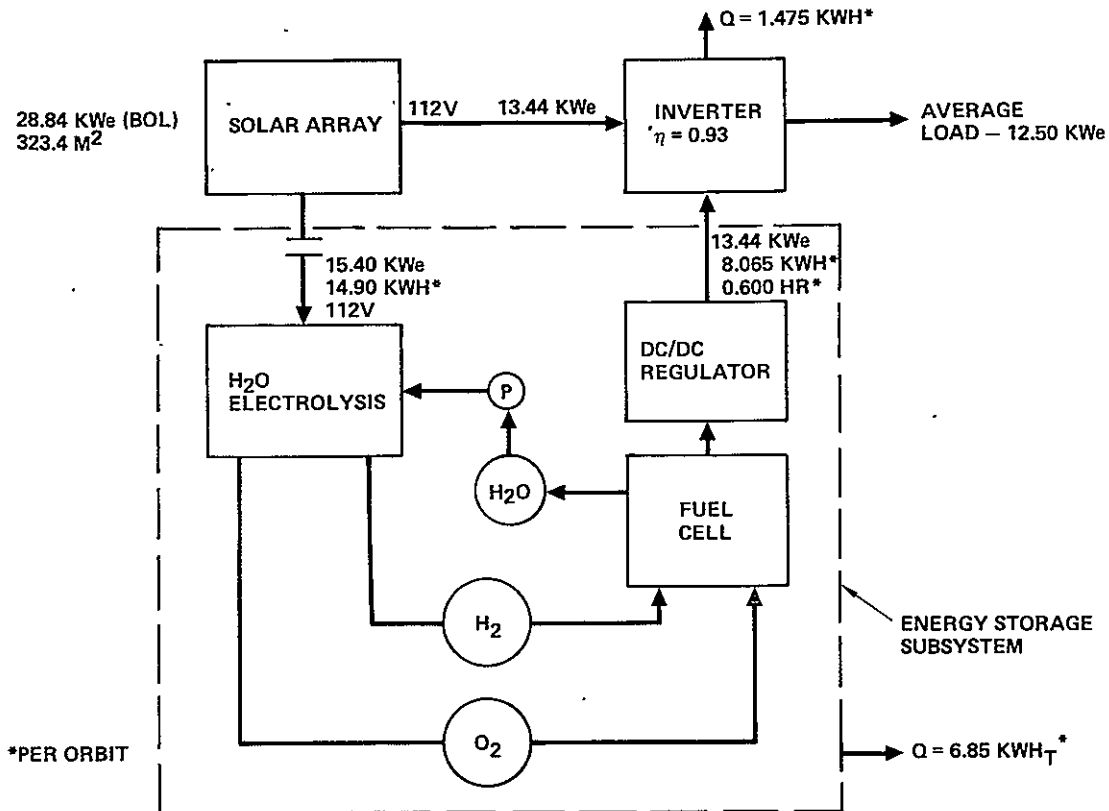


Figure 6-22. Solar Array Regenerative Fuel Cell System (1 of 8, 12.5 kW Each, Fab and Assy)

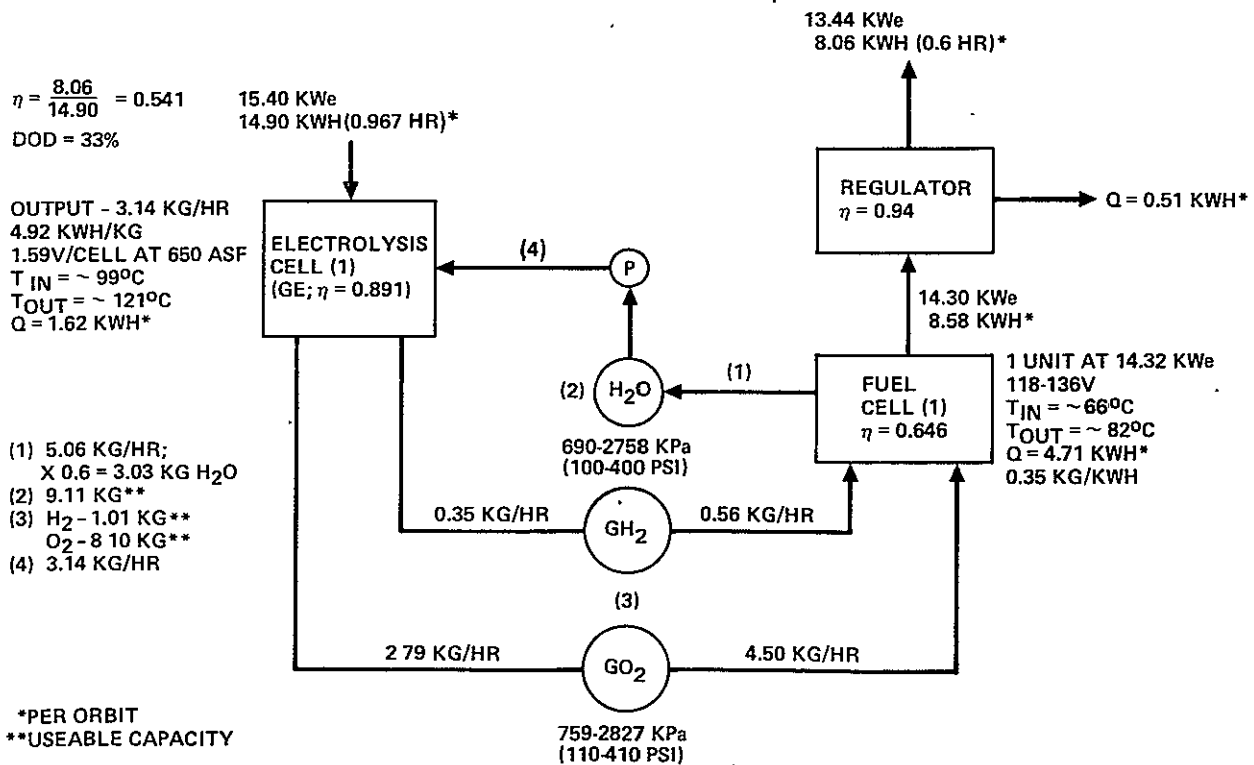


Figure 6-23. Regenerative Fuel Cell Energy Storage (1 of 8)

of 9.11 kg, although the typical nighttime operation only requires 3.03 kg of reactants. Consequently, the system is operating at an effective DOD of 33 percent. The water electrolysis cell is a General Electric unit. The electrolysis cell output is 3.14 kg per hour; its power consumption is 4.92 kW-hr/kg. The unit operates at temperatures in the range of 200-250°F. The overall energy storage efficiency is 54.1 percent.

The life of the RFC system is a very important parameter. The Shuttle fuel cell life is 5,000 hours, with maintenance at 4.5 kWe load, and is limited by voltage degradation with age. Transient voltage performance is also an important factor in life/voltage regulation considerations. Figure 6-24 shows Shuttle voltage regulation performance. The SCB RFC system employs a dc/dc regulator at the output of the fuel cell. This regulator can compensate for degradations in output voltage and transient variations. It is assumed that the effective fuel cell life can be doubled because of this output regulator.

The life of the Shuttle fuel cell is also a function of load, as indicated in Figure 6-25 and Reference 6-8. The Shuttle fuel cell life is rated at an average load of 4.5 kWe. The SCB design point has been selected to be equivalent to operating the Shuttle fuel cell at 3.6 kWe, which increases the life by a factor of 1/0.7 for a 43 percent improvement. Fuel cell life is also a strong function of operating temperature, as shown in Figure 6-26, (Reference 6-9). For example, reducing the operating temperature from 88°C (190°F) to 74°C (165°F) increases the life from 5,000 to 10,000 hours. 74°C is the tentative design point for the SCB application.

Figure 6-27 summarizes the various factors that suggest long life for Shuttle technology fuel cells in the SCB application. The first factor represents an increase to 10,000 hours because of the increased voltage range discussed earlier. The second factor relates fuel cell operation time to mission life — the fuel cell operates only 36 minutes during each 94-minute orbit. Consequently, mission life is 94/36 times the fuel cell operating life requirement. The third life improvement factor relates to operation at reduced loads relative to the Shuttle design point as discussed previously. The last factor results from reduced temperature operation. All of these factors combine to indicate the potential for perhaps eight years of mission life for a fuel

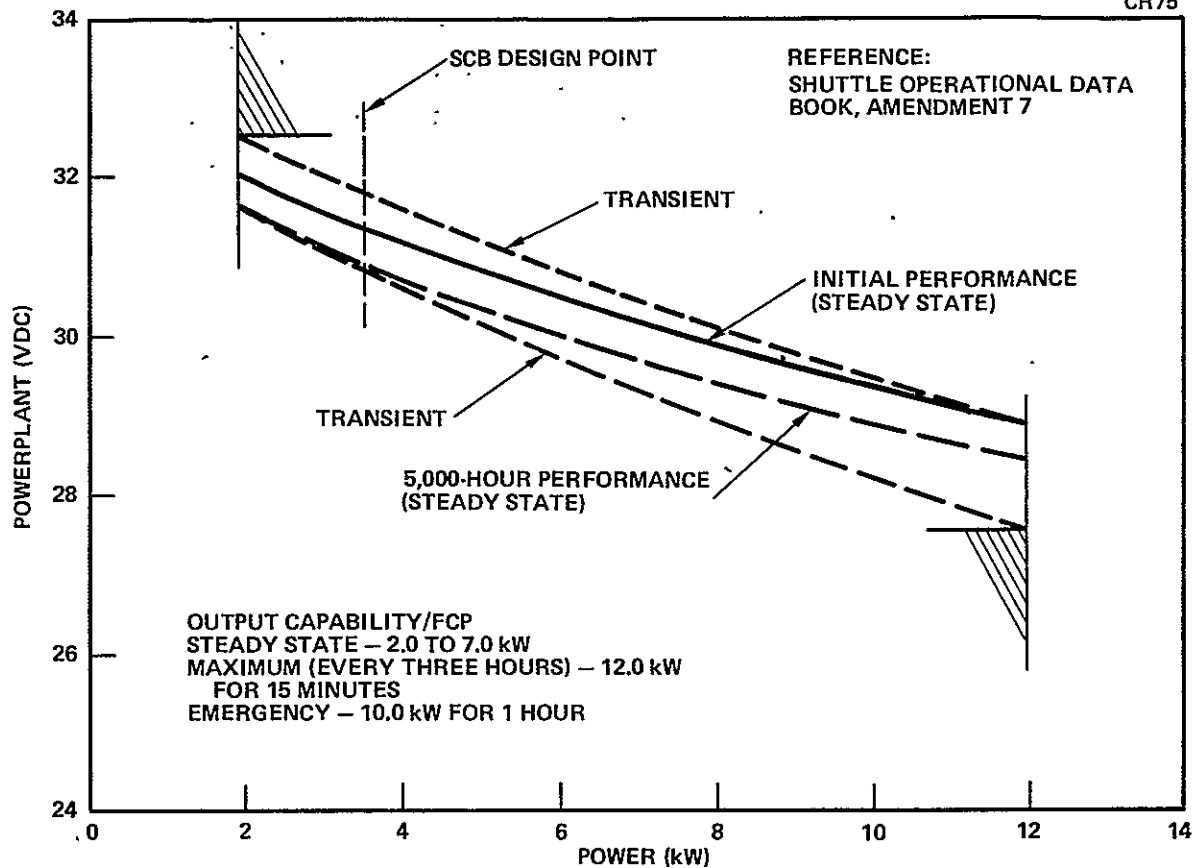


Figure 6-24. Shuttle Fuel Cell Voltage Regulation/Life

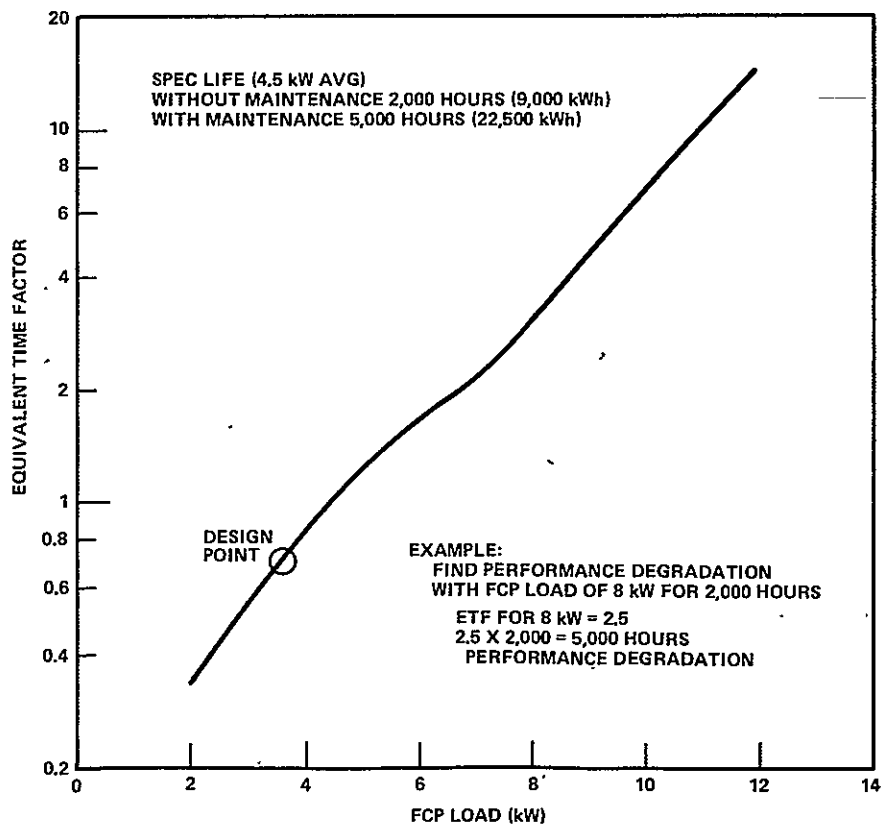


Figure 6-25. Shuttle Fuel Cell Life as a Function of Load



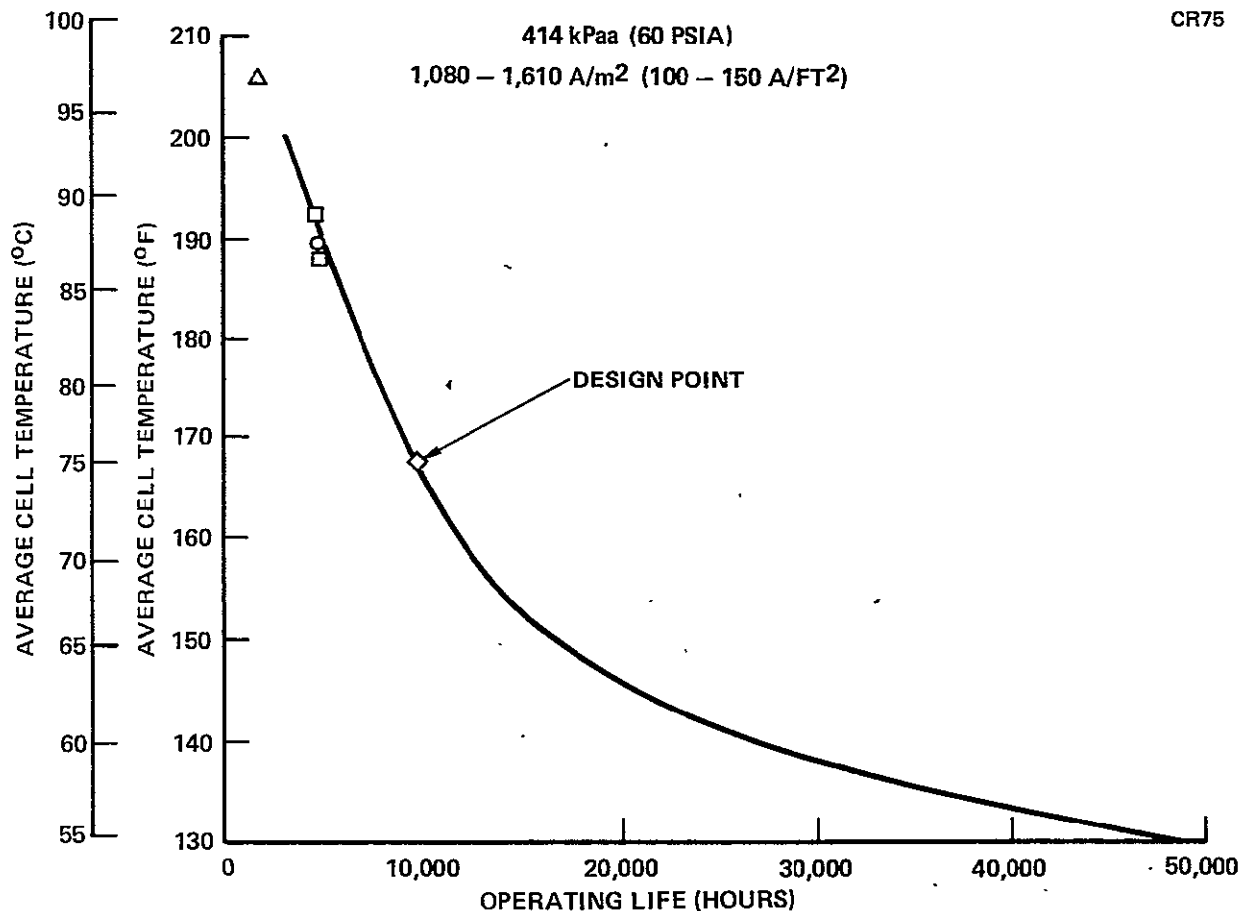


Figure 6-26. Fuel Cell Operating Life as a Function of Temperature

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INCREASED VOLTAGE RANGE    ECLIPSE PORTION OF ORBIT    REDUCED LOAD (4.5 → 3.6 KWe)

$$\frac{10,000 \text{ HR}}{8,760 \text{ HR/YR}} \times \left( \frac{94}{36} \right) \times \frac{1}{0.7} \times 2 = 8.52 \text{ YR MISSION LIFE}$$

REDUCED TEMPERATURE  
(88 → 74°C)  
(190 → 165°F)

Figure 6-27. Shuttle Fuel Cell Life Estimate

cell based on Shuttle technology. Further analysis and testing is required to verify this preliminary estimate. GE fuel cells also have demonstrated long life on small samples and have excellent life potential. They are also a good candidate for SCB application.

The GE water electrolysis cell has demonstrated long life, as shown in Figure 6-28 (20,000 operating hours — Reference 6-10). The electrolysis cell only operates 58 minutes during each 94-minute orbit; consequently, the 20,000 hours of operating life is equivalent to 3.7 years of SCB mission duration. The General Electric fuel cells and water electrolysis cells exhibit little or no degradation with age. Consequently, it is reasonable to expect 5 and perhaps 10 years of mission life with the GE water electrolysis and fuel cells; the SCB design is based on a five year water electrolysis system life.

#### 6.3.6 Energy Wheels

The energy wheel (fly wheel) energy storage system is discussed in this section. The energy wheel system typically consists of a pair of counter-rotating fly wheels, each of which is driven by a motor/generator (Figure 6-29). During the eclipse portion of the SCB orbit, energy is extracted from the energy wheel to power the SCB; this results in reduced energy wheel rotational speed. The kinetic energy (KE) is proportional to the square of the rotational speed ( $\omega^2$ ). Energy is returned to the energy wheel from the solar array via the motor during the sunlight portion of the SCB orbit. Energy wheels may also be used to provide the CMG function in some applications. In this case, the momentum (H) is directly proportional to the rotational speed ( $\omega$ ). The net momentum vector of a pair of counter-rotating wheels must be zero in order not to perturb the SCB orientation.

Figures 6-30 and 6-31 present block diagrams of the energy wheel and energy wheel storage systems. The portion of the total system depicted employs 16 energy wheels (8 pairs); each wheel contains 1.46 kW-hr energy at 45,000 rpm. The minimum practical speed is 22,500 rpm, at which the residual energy is 0.365 kW-hr for each wheel. Reducing the speed from 45,000 to 35,800 rpm provides 49 percent of the available energy; this is the

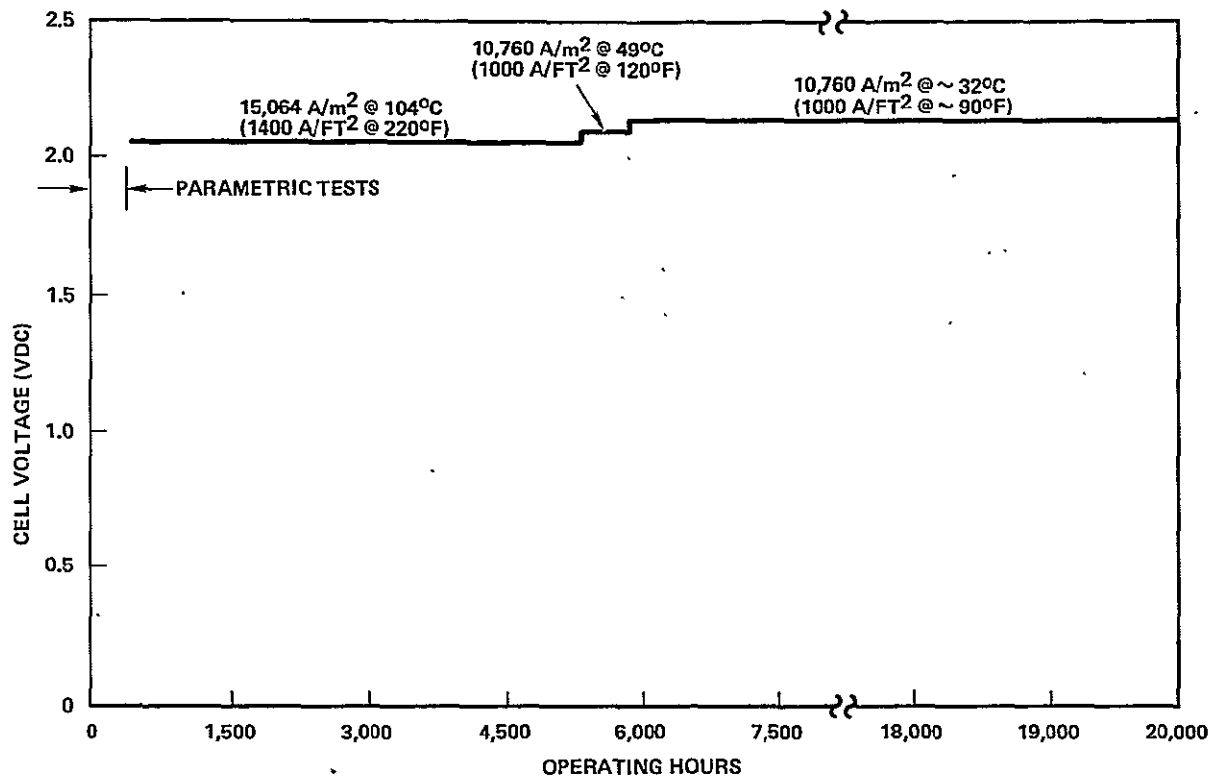
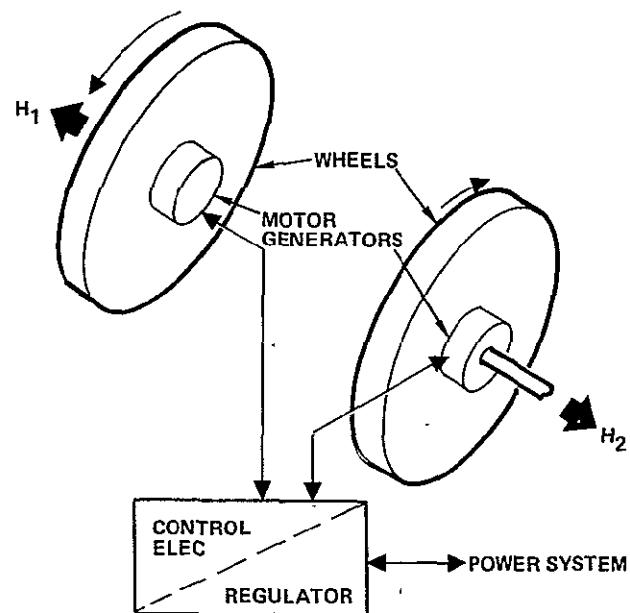


Figure 6-28. SPE Water Electrolysis High Current Density Life Test



$$\left. \begin{array}{l} KE = \frac{1}{2} I \omega^2 \\ H = I \omega \end{array} \right\} \text{FOR PURE ENERGY TRANSFER} \\ \text{TO/FROM THE WHEEL PAIR:} \\ H_1 - H_2 = 0$$

Figure 6-29. Energy Wheel Concept

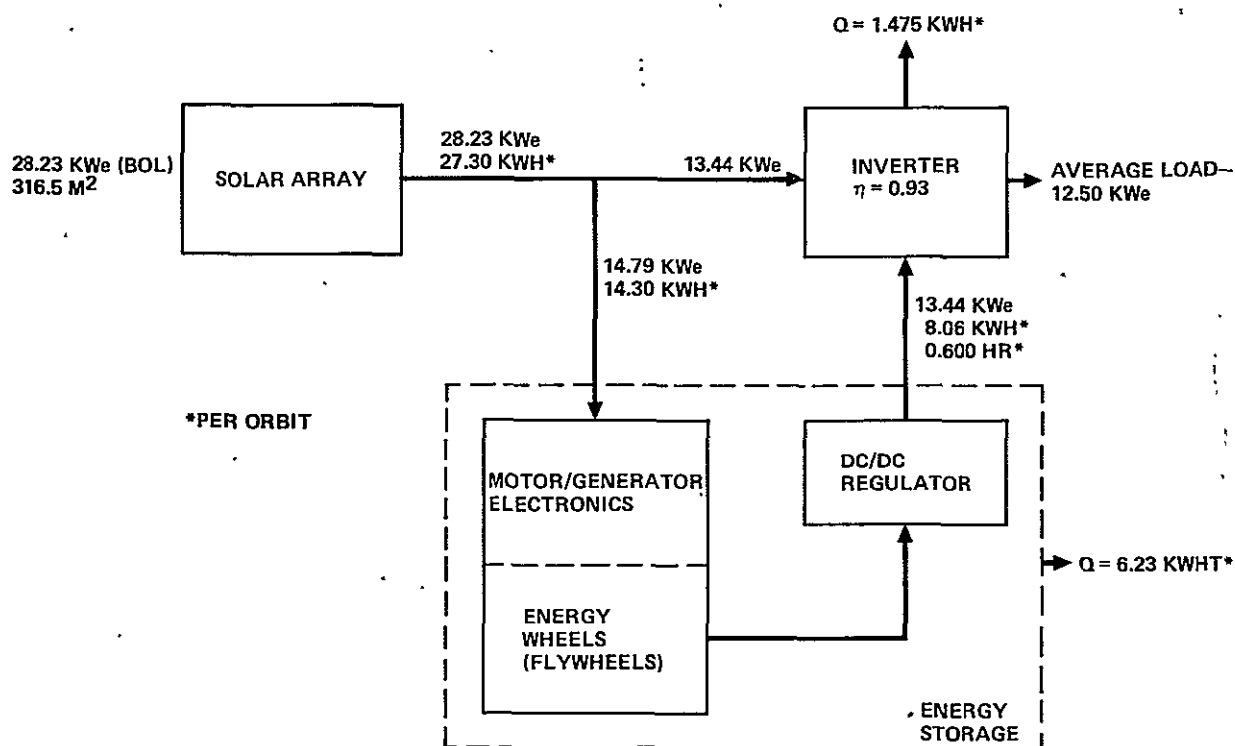
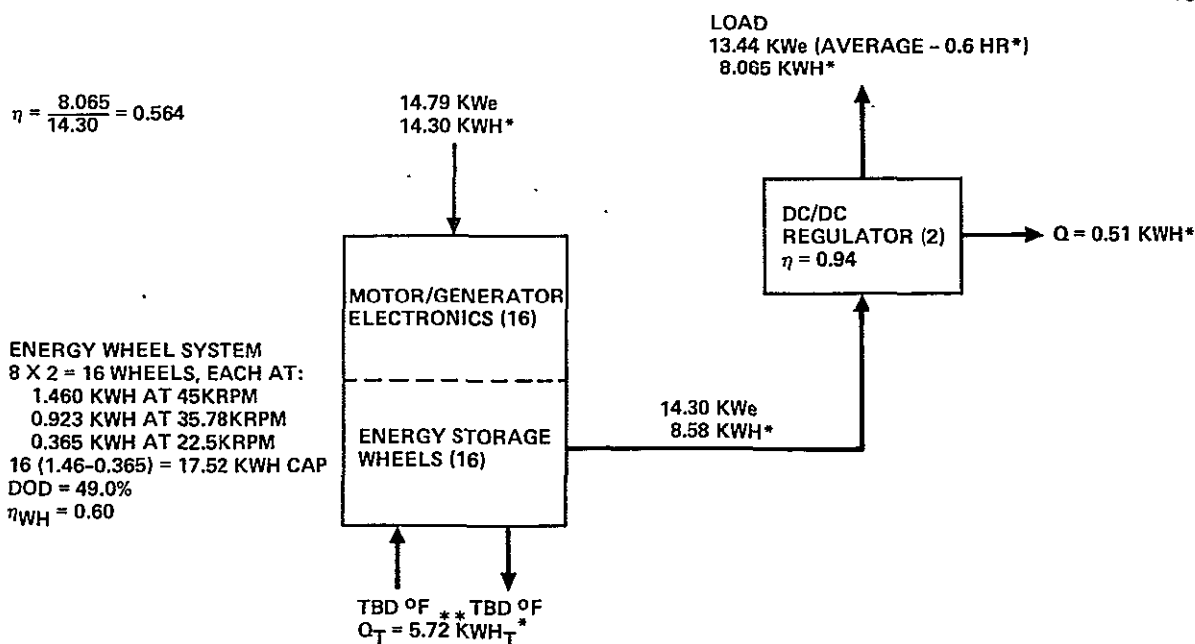


Figure 6-30. Solar Array Energy Wheel System (1 of 8, 12.5 kWe Each, Fab and Assembly)



\*PER ORBIT  
 \*\*LIQUID COOLING PROVISIONS ARE REQUIRED

Figure 6-31. Energy Wheel Energy Storage (1 of 8)

amount of energy typically extracted during the SCB eclipse period. Thus, the energy wheel system operates at 49 percent DOD. This large DOD leaves a marginal amount of reserve energy available for peaks and long-term load averaging. The energy wheel cooling temperatures are TBD, but are expected to be higher than those for the battery systems and are expected to present no significant vehicle integration problem. The overall energy storage efficiency for this system is 56.4 percent which results in a solar array area of 316.5 m<sup>2</sup>.

The preceding discussion was based on steel wheels running on ball bearings, which represent reasonable current technology. An advanced technology energy wheel system would employ composite wheels running on magnetic bearings. Table 6-7 summarizes the key characteristics of these two energy wheel options. The weights shown do not include containment in the event of wheel rupture. Also, the 49 to 51 percent DOD is marginal with respect to reserve energy capability, as noted earlier. Consequently, the weights shown are probably on the low side.

The life of the steel wheel/ball bearing energy wheel system is expected to grow with program maturity in much the same manner as aircraft jet engine time between overhauls (TBO) grows. Aircraft jet engine TBO is typically 1,000 hours at the initiation of service and grows to 10,000 to 20,000 hours after 5 to 10 years in service. Consequently, an expected life of 1 to 2 years is predicted for the energy wheel system at least early in the SCB program.

Figure 6-32 shows the relationship of energy wheel speed and diameter for three types of energy wheels. The constant stress wheel exhibits a diameter of approximately 0.35 meters at 50,000 rpm, which is believed to be a reasonable upper speed. Therefore, it is necessary to employ a large number of relatively small wheels, which aggravates the production and replacement hardware operational costs. A summary of the characteristics of Rockwell International Integrated Power/Attitude Control System (IPACS) energy wheels is presented in Table 6-8 (Reference 6-11). The steel wheels with ball bearings are considered most appropriate for the SCB application, because of their technology status.

Table 6-7

**ENERGY WHEEL ENERGY STORAGE SYSTEM CHARACTERISTICS**  
(8-Module Total, 100 kWe Average)

|                                                            | Steel Wheels<br>(Ball Bearings) | Composite Wheels<br>(Mag Bearings) |
|------------------------------------------------------------|---------------------------------|------------------------------------|
| Output Power, kWe                                          | 107.5                           | 107.5                              |
| Average Output Energy, kW-hr                               | 64.52                           | 64.52                              |
| Average Input Energy, kW-hr                                | 114.4                           | 98.04                              |
| Energy Efficiency, %                                       | 56.4                            | 65.8                               |
| Number of Wheels                                           | 128                             | 48                                 |
| Nominal DOD, %                                             | 49.0                            | 51.1                               |
| Expected Life, Years                                       | ~1-2                            | ~1-2                               |
| Energy Wheel Specific Energy<br>(Nominal/Maximum), W-hr/kg | 8.75/17.9                       | 10.9/21.3                          |
| Weight, kg                                                 | 8,024                           | 6,500                              |
| • Energy storage wheels<br>(128/48) <sup>(1)</sup>         | (7,256)                         | (6,095)                            |
| • Motor/generator<br>electronics (128/48) <sup>(1)</sup>   | ( 581)                          | ( 218)                             |
| • Controller (8)                                           | ( 9.1)                          | ( 9.1)                             |
| • Regulators (16/16) <sup>(1)</sup>                        | ( 178)                          | ( 178)                             |

<sup>(1)</sup>Number of units — (steel wheel/composite wheel)

### 6.3.7 Radiator Analysis

The radiator analysis discussed in this section consists of radiator sizing tasks for the SCB power platform system with NiCd batteries and the deployable solar array with RFC for energy storage. Both designs dissipate the bulk of their waste heat on the shade side of the orbit during energy storage device discharge. Shade side dissipation was therefore considered the design point.

### Requirements

Requirements differ for the two configurations because of (1) radiator configuration, (2) energy storage device cooling temperatures, and (3) amount of heat dissipated. These differences can be seen in Table 6-9. The NiCd

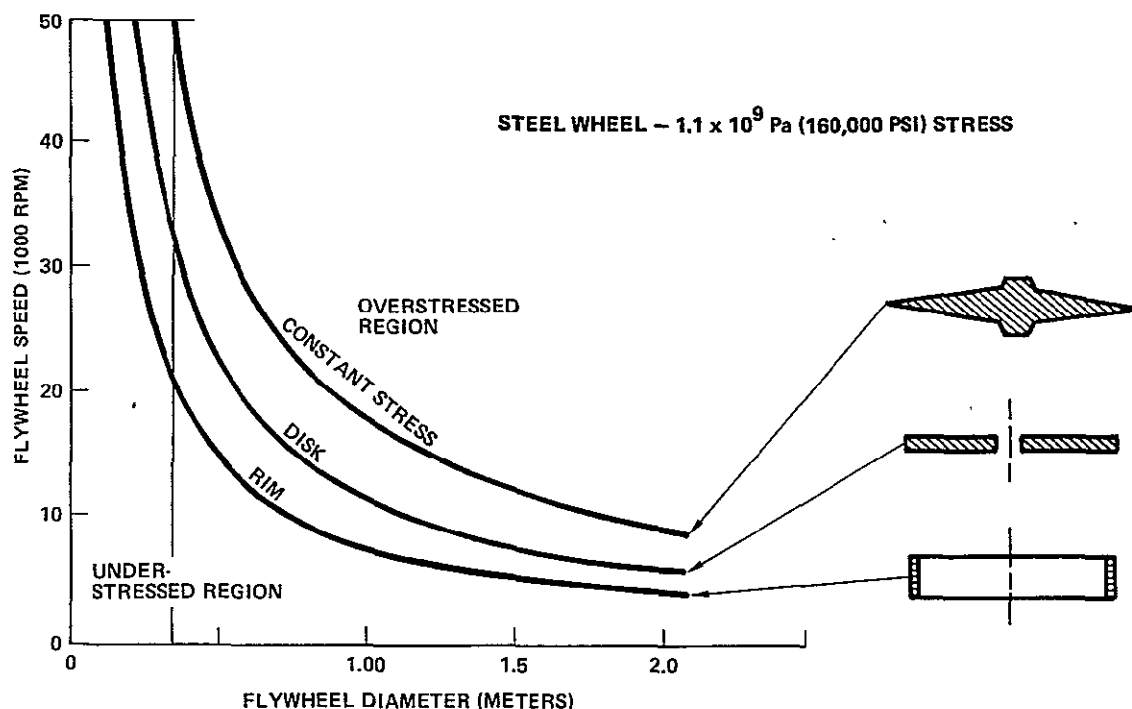


Figure 6-32. Maximum Operating Speeds, Isotropic Rotor

Table 6-8  
IPACS WHEELS

|                                    | Ball Bearing-<br>Steel  | Mag Bearing-<br>Composite |
|------------------------------------|-------------------------|---------------------------|
| Wheel Weight, kg                   | 44.0                    | 93.9                      |
| Wheel Diameter, m                  | 0.378                   | 0.579                     |
| Wheel Speed                        | 45,000 rpm              | 35,000 rpm                |
| Wheel Energy                       | 1460 W-hr at 45K rpm    | 3730 W-hr at 35K rpm      |
| Wheel Momentum, N-m-sec            | 2229 N-m-sec at 45K rpm | 5690 N-m-sec at 35K rpm   |
| Average Bearing and Windage Losses | 80 watts                | 33 watts                  |
| Package Weight*, kg                | 56.7                    | 127                       |

\*Estimated - Assumes removal of gimbals and torquer

Table 6-9  
ENERGY STORAGE COOLING REQUIREMENTS

| Requirement                       | Regenerative Fuel Cell | NiCd Batteries                      |
|-----------------------------------|------------------------|-------------------------------------|
| Cooling load                      | 74.36 kW               | 40 kW                               |
| Inlet temperature                 | 65.6°C (150°F)         | 10°C (50°F)                         |
| Outlet temperature                | 82.2°C (180°F)         | 20°C (65°F)                         |
| Radiator type                     | Deployed cruciform     | Flat fixed and deployed             |
| Design point beta angle           | 0°                     | 0°                                  |
| Radiator design point orientation | ⊥ to earth surface     | ⊥ to solar arrays and earth surface |

battery cooling load is lower than the RFC because of higher efficiency and part of its waste heat (inefficiency) is dissipated on the sun portion of the orbit.

#### Analytical Approach

Both radiator configurations analyzed consisted of flat surfaces with heat rejection either from one side for fixed radiators or both sides for deployed radiators. A simplified closed-form solution was used which is commonly known as the sink temperature approach. It is discussed in Reference 6-12.

The heat rejection design point for both energy storage concepts occurs on the shade side of the orbit where the greatest amount of waste heat occurs. The heat rejection capability of the radiators does not vary greatly on the shade side since the heat influx from the environment is constant, consisting only of earth IR. Variations in performance are therefore only influenced by radiator orientation and heat flux from nearby vehicle elements. These effects are expected to be small and difficult to account for in the simplified analytical method described above. Therefore, a single point was analyzed on the shade side and vehicle element effects were neglected.

#### Results

Figures 6-33 and 6-34 present the analytical results in terms of effective radiator area required as a function of cooling temperature. Both figures



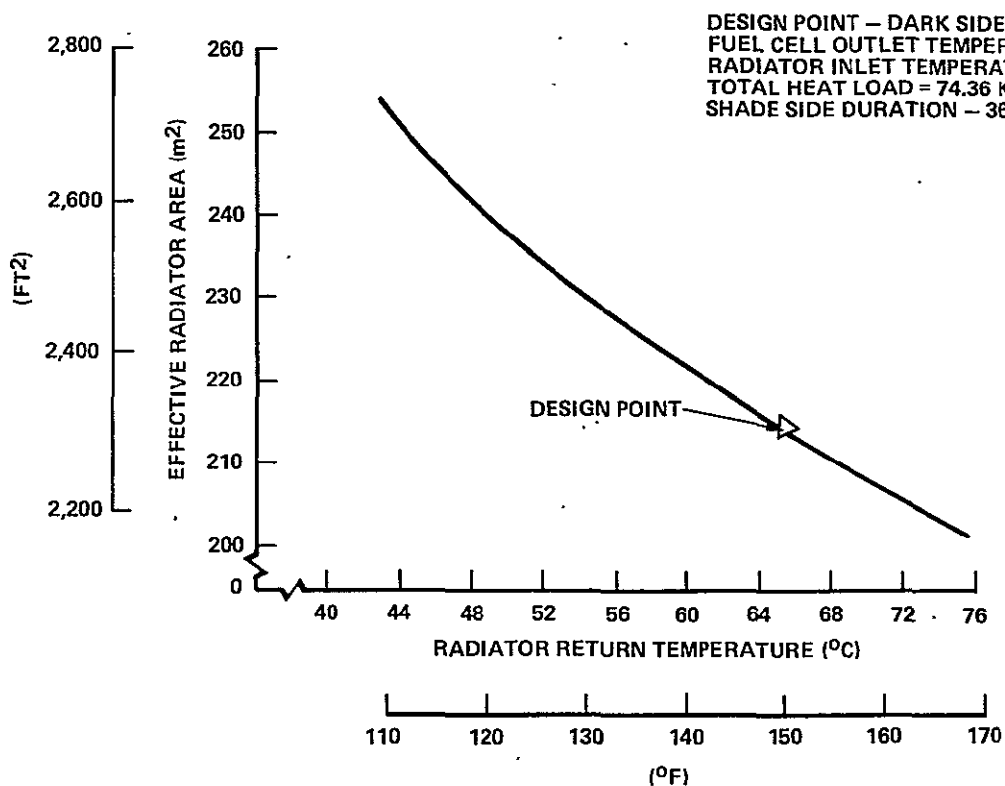


Figure 6-33. Deployable Solar Array Radiator Performance, Fuel Cell Energy Storage

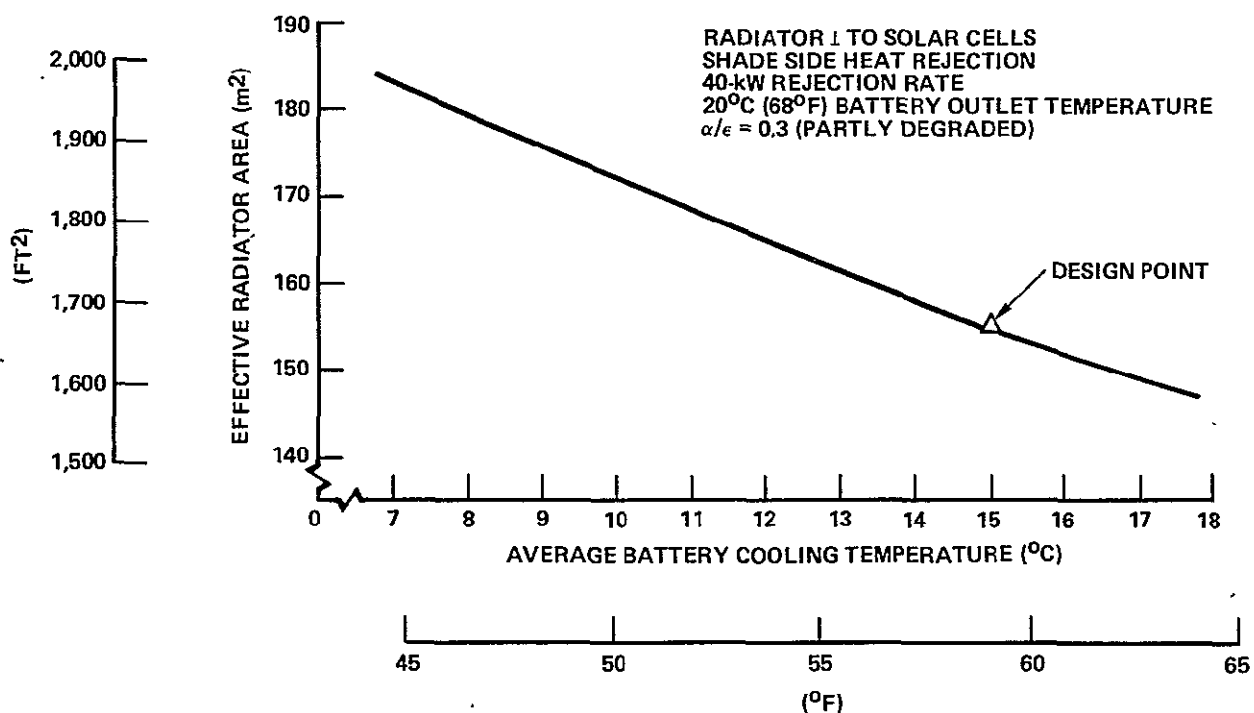


Figure 6-34. Power Platform Radiator Performance, Battery Storage

assume a partly degraded radiator coating ( $\alpha_s/\epsilon_t = 0.3$ ). Figure 6-33 shows that about 215 m<sup>2</sup> (2312 ft<sup>2</sup>) of effective radiator area is required to reject 74.36 kW of waste heat for RFC energy storage. This corresponds to about 0.345 kW/m<sup>2</sup> (0.032 kW/ft<sup>2</sup>) with an inlet temperature of 86.3°C (187°F) and an outlet of 65.6°C (150°F). The resulting average fuel cell temperature is 74°C (165°F).

Figure 6-34 gives effective radiator requirements for NiCd battery energy storage. About 155 m<sup>2</sup> (1668 ft<sup>2</sup>) are required to reject the 40 kW of waste heat load for an average battery cooling temperature of 15°C (59°F). A total coolant temperature difference of 10°C (18°F) was assumed. This heat rejection corresponds to about 0.258 kW/m<sup>2</sup> (0.024 kW/ft<sup>2</sup>), which is about 30 percent lower than for the regenerative fuel cell energy storage concept. This difference is due primarily to the lower cooling temperatures required for NiCd batteries.

Both figures show that radiator requirements appear nearly linear with temperature. This is because a relatively small temperature range is shown on the figures and the sink temperature is quite low; -88°C (-127°F) for the design point. This low sink temperature results from the favorable radiator orientation for this particular design, wherein the radiator flat area is perpendicular to the earth's surface. As a result, only a limited amount of earth IR is absorbed. Since radiator area becomes large as the radiator outlet fluid temperature approaches the sink temperature, less favorable radiator orientations could result in much larger required areas. This is particularly true for the battery concept, since a much lower cooling temperature is required.

#### 6.4 ALTERNATE BASELINE SYSTEM (DEPLOYABLE SOLAR ARRAY)

The Deployed Solar Array system, which was shown earlier in the right-hand column of Table 6-1, is discussed in this section. This work was accomplished as part of the Task 10 effort, to define an alternative to the fabricated and assembled baseline power platform coupled with an advanced technology energy storage system (RFC's). The system features a deployable solar array based on the use of multiple SEP units, with current-technology solar

cells and RFC energy storage. Two advanced-technology, low-cost silicon solar cell concepts are compared with the current SEP solar cell technology. The operations/buildup example discussed in Section 6.4.2 is based on the Shuttle sortie/free flyer buildup mode.

#### 6.4.1 System Description

An overview of the SCB powered by a deployable solar array is presented in Figure 6-35. The solar array is gimballed about both the X and Y axes, the gimbal being located at the "array drive assembly" indicated on the figure. Because area available for the radiator on the energy storage module is limited, the required 215-m<sup>2</sup> radiator is attached to the RC pod support structure. The 215-m<sup>2</sup> radiator area is an effective area, assuming heat transfer from both sides of the radiator flat plates. The SCB buildup sequence is indicated in the box at the left of the figure. The details of this buildup will be discussed subsequently. Figure 6-36 shows more detail of the solar array, the energy storage module, and the related support structure. The solar array consists of 20 SEP units, clustered in groups of five, attached to the support beams. A telescoping tunnel assembly attaches the energy storage module to the two-axis gimbal device, which is the MDAC Phase B Space Station design. The two-axis gimbal system is completely maintainable in a shirtsleeve environment, as is equipment located in the energy storage module. The energy storage module contains the fuel cells, electrolysis cells, power conditioning equipment, and displays and controls. The O<sub>2</sub> and H<sub>2</sub> gas tanks for the RFC system are located on the exterior of the module, as noted. The system characteristics are listed in the box of Figure 6-36. The electrical power output at BOL is 100 kWe average; the corresponding solar array output is 230.7 kWe. The weight of 14,588 kg includes the radiator, which is not shown in this figure. The solar array output is less than output of the baseline power platform array discussed previously, because the deployable solar array is always solar oriented, despite the increased output requirement resulting from the lower efficiency of the fuel cell system selected for the deployable array system. The two-axis gimbal system, as contrasted to the one-axis system discussed earlier, provides: (1) complete SCB orientation flexibility (e.g., inertial orientation); (2) lower attitude control propellant requirements; (3) a slightly smaller solar array for equivalent energy storage system efficiency; these benefits

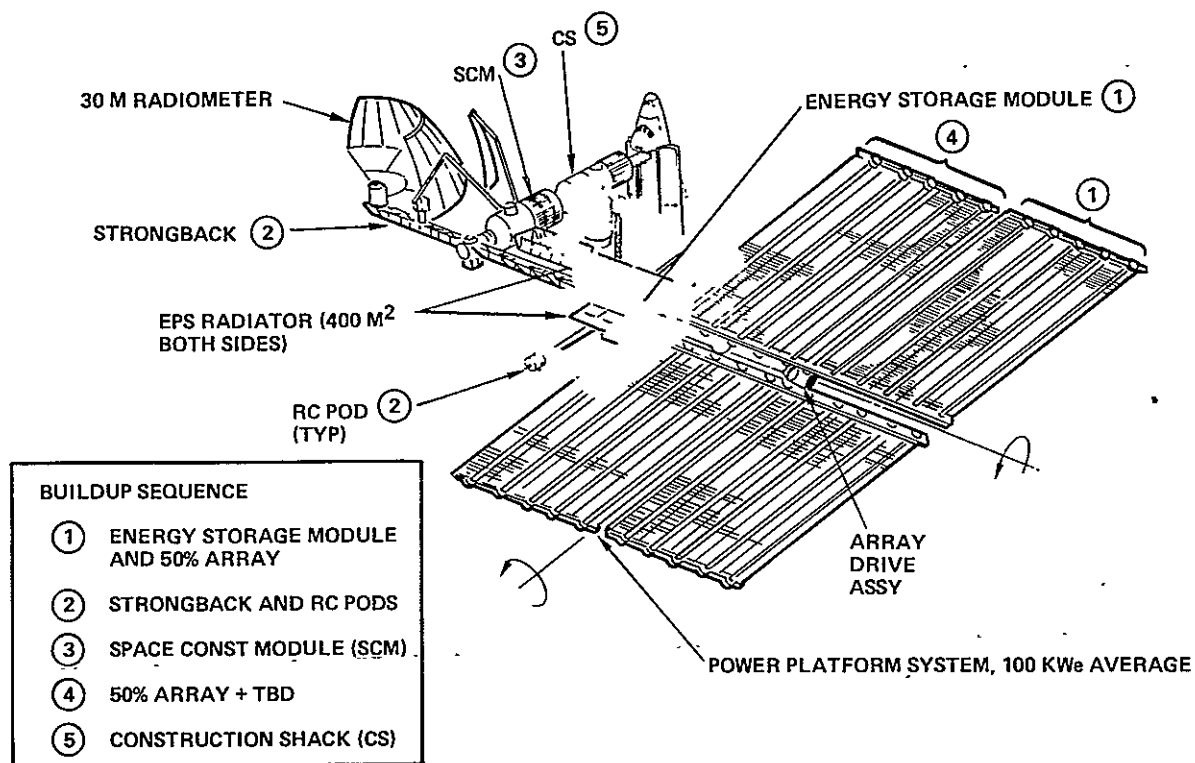


Figure 6-35. SCB Configuration Deployable Solar Array

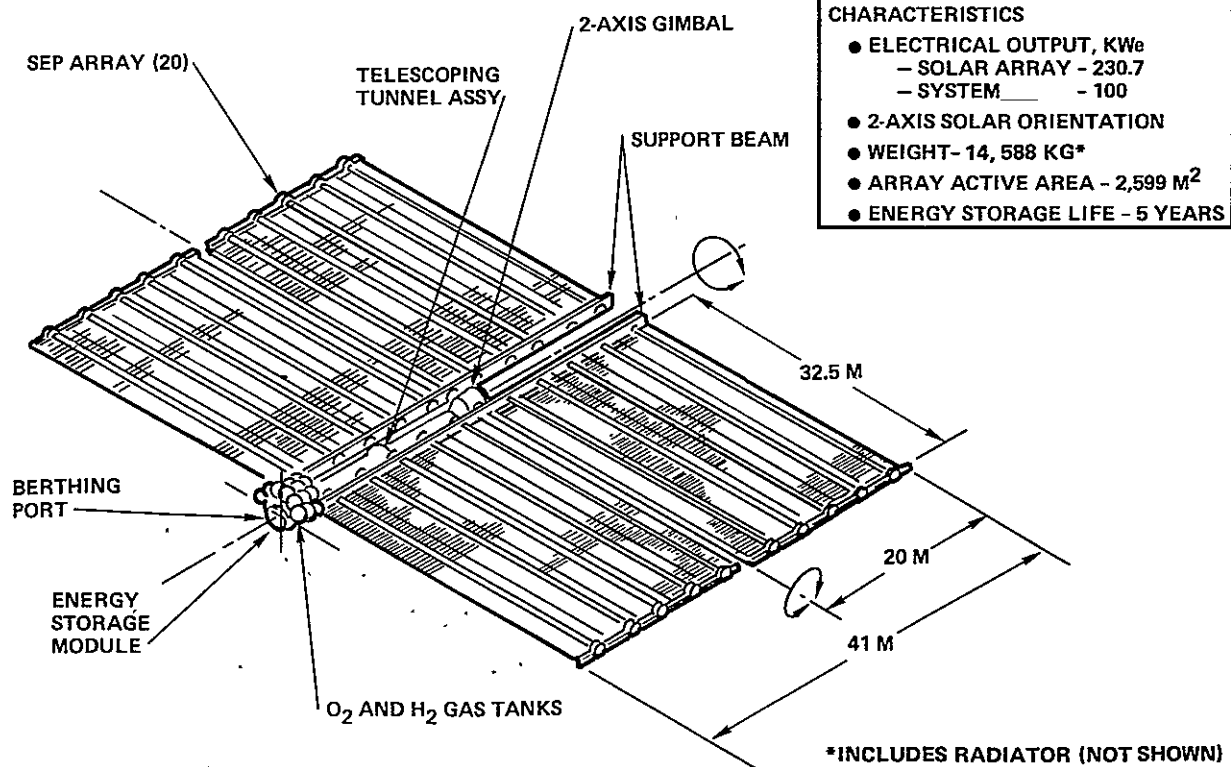


Figure 6-36. SCB Power Platform, Deployable Solar Array

are at the expense of a heavier, more complex, and more expensive orientation system.

The SEP solar arrays employed on the deployable solar array have been under development for a number of years and indeed are an outgrowth of the Phase B Space Station solar array, which had many similarities to the array depicted in Figure 6-36. MDAC has elected to use an array slightly longer than the SEP array (32.5 meters, rather than 31.6 meters).

The initial SCB deployable array utilizes ten of the baseline SEP arrays packaged for launch as shown in Figure 6-37. Three arrays are mounted on each of two 12-meter-long support beams and two arrays are mounted on each of two 8-meter-long support beams. The 12-meter-long support beam incorporates the mounting flange, which is mated to the gimbal mechanism incorporated in the array drive and orientation installation. The array drive and orientation turret is mounted on the end of a 1.5-meter-diameter telescoping tunnel and incorporates a 1.0-meter hatch for maintenance.

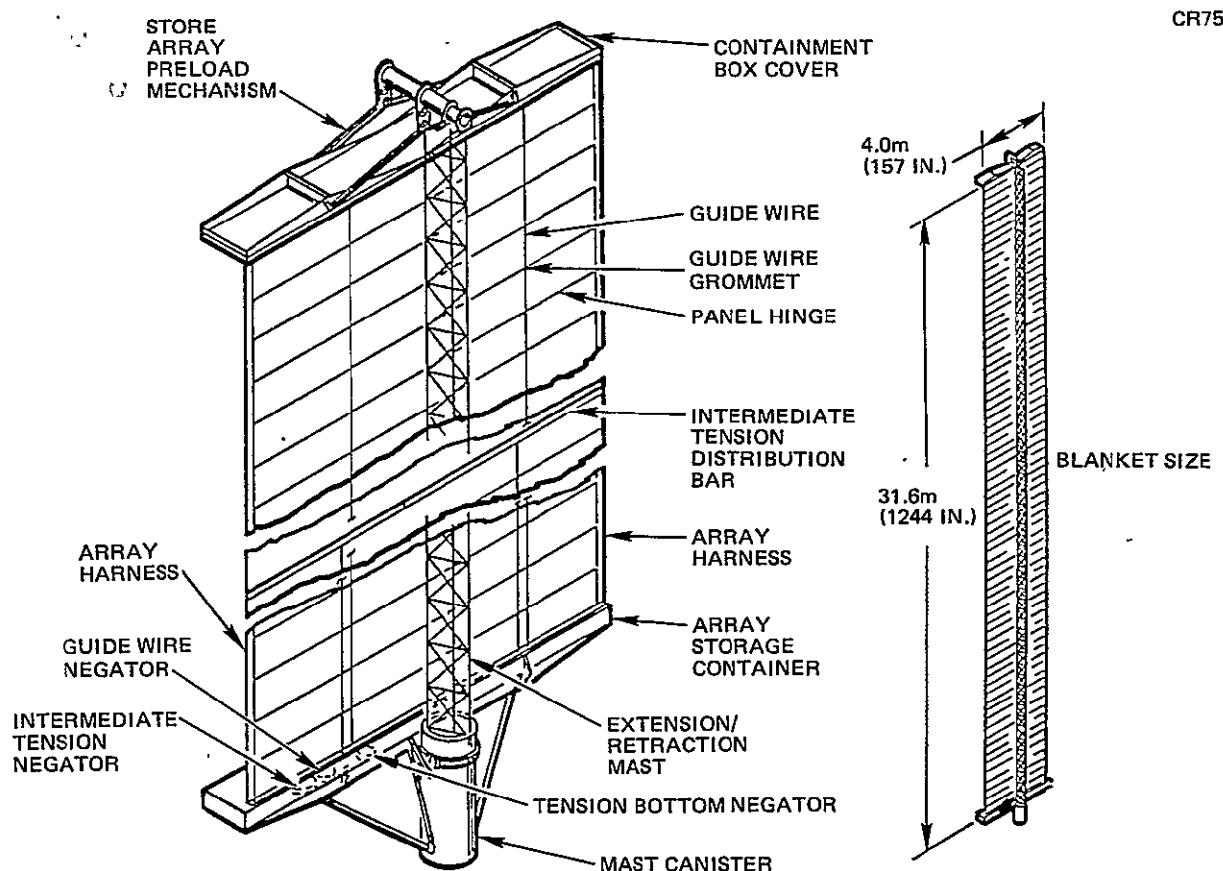


Figure 6-37. SEP Solar Array Wing

nance purposes. Electronic components are housed in a 3.0-meter-diameter module, which incorporates docking/berthing capabilities with the Orbiter docking module and the SCB strongback. The fuel cells, electrolysis cells, and the power conditioning equipment are packaged in the interior of the module, with the  $\text{GO}_2$  and  $\text{GH}_2$  tanks packaged on the external surface.

The initial launch package utilizes an envelope with a 4.42-meter diameter and 16 meters in length. (See Figure 6-38).

The major components of the deployable solar array are mounted on a tubular-type, V-shaped, deployable launch pallet. The primary loads are transferred to the Orbiter through a four-point retention system. The  $\pm X$  and  $\pm Z$  loads will be transferred to the Orbiter longeron sill through two trunnions at approximately Station 1175.20 and one trunnion at approximately Station 715.0. The  $Y$  loads will be transferred through the cargo bay keel fitting at approximately station 1175.20. Each of these load retention stations is adjacent to the primary load-carrying structure of the deployable array system. Secondary removable structure supports the 12-meter-long array support beams from the tunnel assembly and transfers the launch loads to the assembly which is supported by the pallet. The 8-meter-long array support beams are mounted directly to the pallet support system. Following rotation and berthing of the launch package, the pallet serves as a holding fixture for assembly of the array system components. Following completion of array assembly, the pallet is returned to the cargo bay.

#### 6.4.2 Orbital Buildup Operations

On orbit, the launch package is rotated out of the cargo bay on the PIDA and berthed at the Orbiter docking module with the RMS. The RMS is then used to mate the mounting flange incorporated in the 8-meter-long support beam with the gimbal fitting on the orientation turret. The telescoped tunnel is then extended, and the 12-meter-long support beam section is attached ("Attach Here," Figure 6-39) to the end of the short assembly completing the initial deployed array. A deployed array system, shown in Figure 6-40, contains a limited communications/navigation system and RCS, which permits it to perform as a free flyer between Orbiter visits.

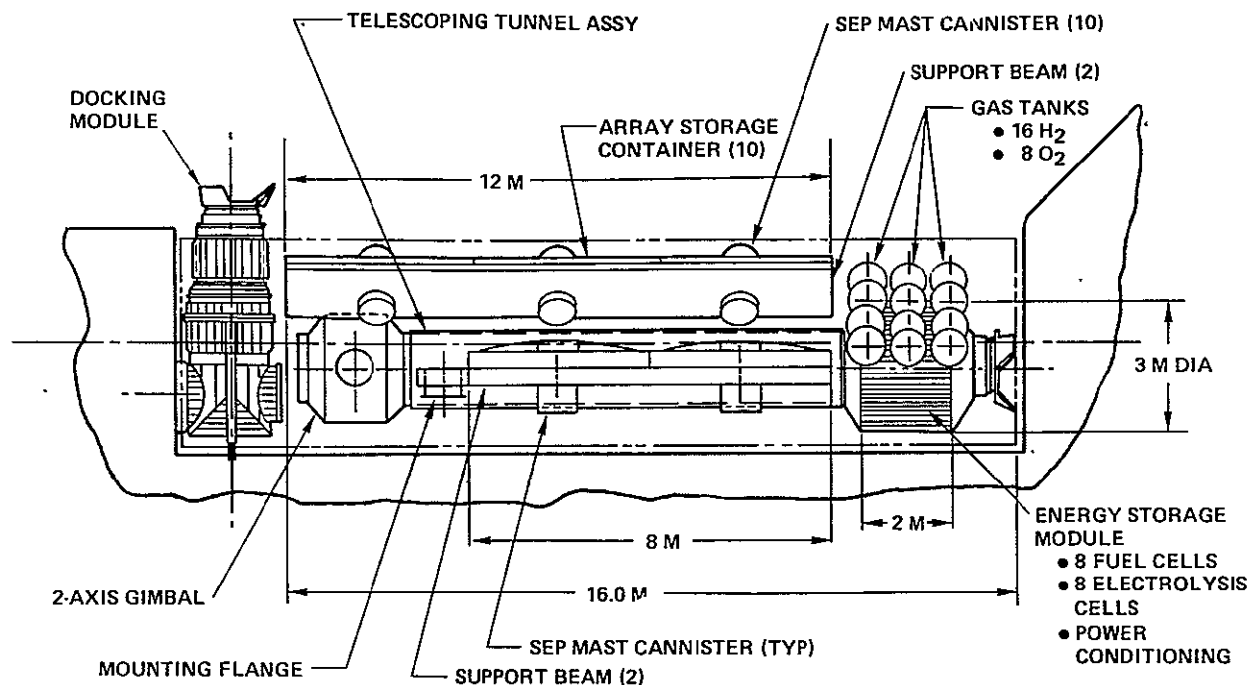


Figure 6-38. SCB Deployable Array Launch Configuration

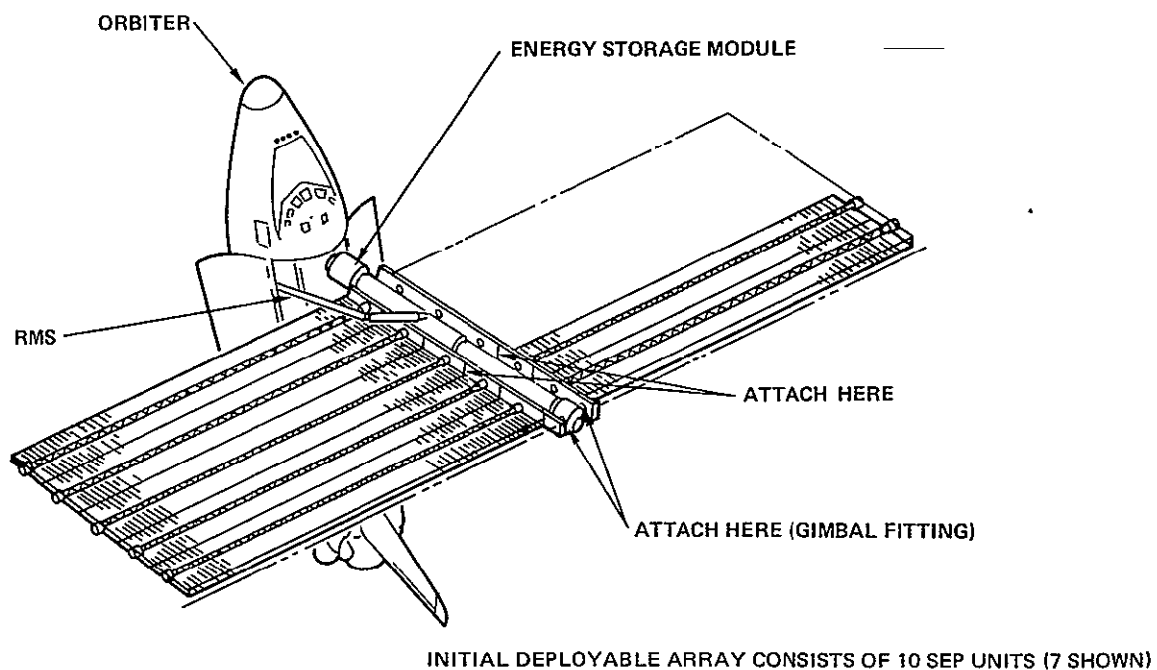


Figure 6-39. Array Delivered and Assembled Using the Orbiter RMS

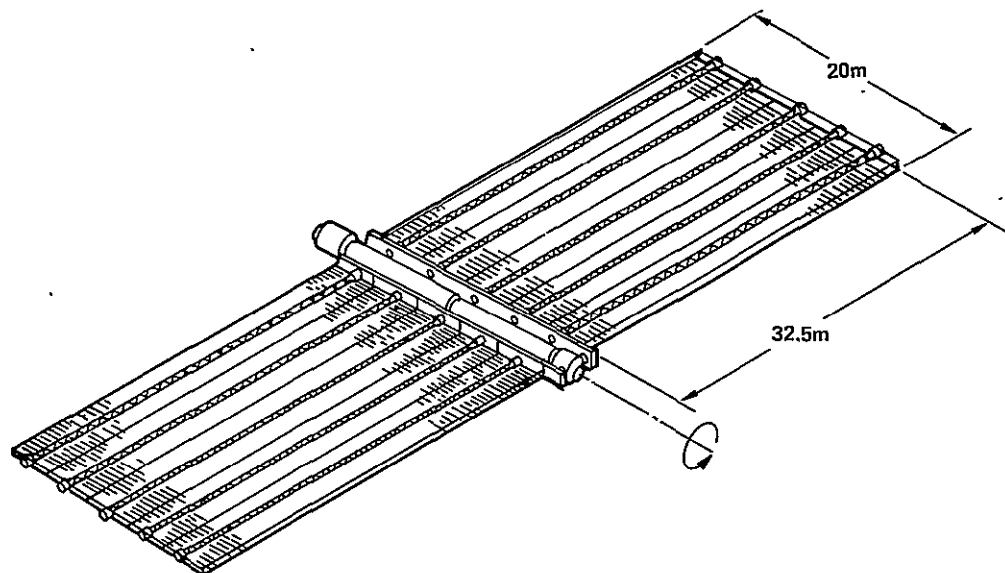


Figure 6-40. Deployable Array System in Free-Flyer Mode

CR75

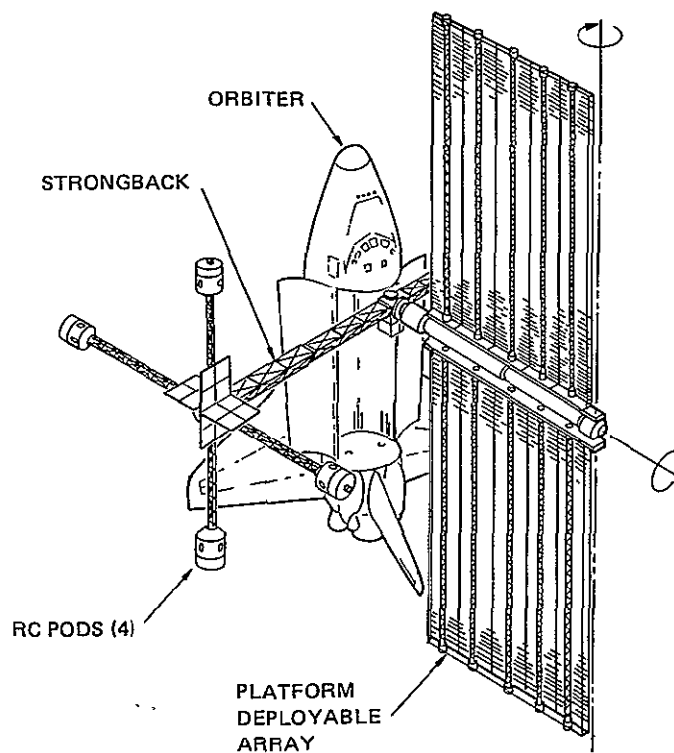


Figure 6-41. Strongback Assembly and Berthing to Deployable Array System



After the solar arrays are deployed and operational integrity has been verified, the array system is released and is left in a nominally quiescent state until scheduled launch of the strongback. Before the Orbiter docks with the array system, the RMS removes the folded truss beam strongback from the Orbiter cargo bay and berths the core structure to the Orbiter docking module. Each of the triangular truss beams is rotated and locked into place, and the reaction control system (RCS) pods and space radiator panels are installed. Following completion of the strongback assembly, the Orbiter and strongback are docked to the array system resulting in the orbital configuration shown in Figure 6-41. After the strongback has been docked, the total assembly is released to await launch of the Space Construction Module (SCM). Following docking operations, the SCM is deployed from the cargo bay by the PIDA. The RMS removes the space crane components from their launch position and assembles them on the SCM in the operational configuration. With the Orbiter docked to the SCM, the solar arrays are retracted and the space crane relocates the entire assembly to the X axis of the strongback, as shown in Figure 6-42. Following berthing verification, the solar arrays are deployed and the Orbiter returns to earth to obtain additional array components.

Ten additional SEP array assemblies are launched: following docking operations of the Orbiter to the SCM, the solar array installation is rotated and the telescoping tunnel retracted in order to maintain the assembly operation within the capability of the space crane. The solar array beam assemblies are moved by the space crane from the Orbiter and attached to the system. As each section of the support beam assembly is completed, the telescoping tunnel is extended until all the SEP array assemblies are installed.

The ten SEP arrays require a large annular volume of the cargo bay, although they utilize only a small fraction of the Orbiter payload capability. A full Shuttle launch has been charged for this launch in the cost evaluations, although it is likely that other cargo can be found to help defray this expense.

Immediately following the completion of the deployable array system, the Construction Shack (CS) is launched into orbit. After completion of docking operations, the CS is removed by the space crane and berthed to a radial

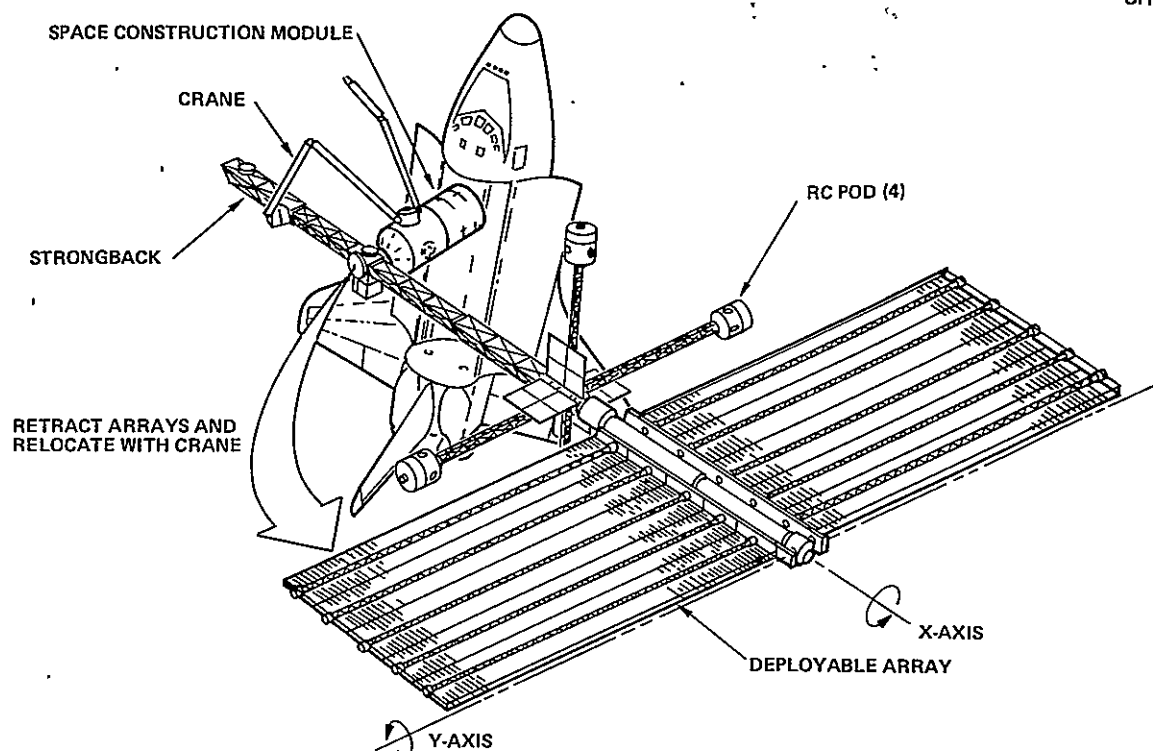


Figure 6-42. The Orbiter Delivers the SCM and Crane

port on the SCM. Following crew transfer, the Orbiter is undocked, and the CS is relocated along the Y-axis. At this time, the SCB is fully assembled, activated, manned, and capable of initiating routine operations in a continuous manned mode of operation with Orbiter support. The resulting on-orbit configuration is shown in Figure 6-35.

#### 6.4.3 Solar Cell System Options

Three solar cell system options were evaluated, as shown in Table 6-10. The first option is the standard SEP solar cell. The second is a high-efficiency, low-cost solar cell. The third option is a low-cost moderate-efficiency solar cell. The two advance technology options (second and third) are under development by NASA. The high-efficiency cell (14 percent) is expected to cost \$50 per watt at 28°C and AMO. The \$50-per-watt number is shown in the center column under left-hand-column entry, "Array Recurring Cost, \$/W: \$/Cell". The \$50 per watt rate results in a cost of \$23.6 per cell. The high-efficiency and low-cost advance-technology cells have an area of 25 cm<sup>2</sup> as contrasted to 8 cm<sup>2</sup> for the SEP solar cell. All of the solar cells are 200 microns thick (0.008 inch). The cell output power (W/cell) is given

Table 6-10  
SOLAR CELL SYSTEM OPTIONS\*

|                                    | SEP                    | High Efficiency          | Low Cost                 |
|------------------------------------|------------------------|--------------------------|--------------------------|
| Bare Cell Efficiency, %            | 11.1                   | 14.0                     | 12.0                     |
| Cell Area, cm <sup>2</sup>         | 8.088                  | 25                       | 25                       |
| Cell Power at 28°C/80°C, W/Cell    | 0.121/0.0927           | 0.472/0.361              | 0.404/0.310              |
| Total Cell Quantity, Cells (80°C)  | 2.58 x 10 <sup>6</sup> | 0.664 x 10 <sup>6</sup>  | 0.773 x 10 <sup>6</sup>  |
| Array Recurring Cost, \$/W:\$/Cell |                        |                          |                          |
| ● Cells (28°C, reference)          | 90.9 : 11              | 50 : 23.6                | 5 : 2.0                  |
| ● Cells (80°C)                     | 119 : 11               | 65.3 : 23.6              | 6.5 : 2.0                |
| ● Cell covering (80°C)             | 64.7 : 6               | 33.2 : 12 <sup>(1)</sup> | 38.7 : 12 <sup>(1)</sup> |
| Cell Assembly Subtotal             | 183.7 : 17             | 98.5 : 35.6              | 45.2 : 14.0              |
| ● Blanket/system assembly (80°C)   | 122.5 : 11.4           | 77 : 27.8 <sup>(2)</sup> | 80 : 24.8 <sup>(2)</sup> |
| Total (80°C)                       | 306.2 : 28.4           | 175.5 : 63.4             | 125.2 : 38.8             |
| Array Area, (80°C), m <sup>2</sup> | 2,599                  | 2,061                    | 2,404                    |

\* Assumes deployed array—Shuttle/Free-Flyer; 230.7 kWe array (239.6 kWe at cells)

(1) Cell unit cost is 2 x SEP

(2) SEP extrapolated; average of 122.5 \$/W and 11.4 \$/cell

in the "Cell Power" entry of the left-hand column for both 28°C and 80°C operation.

The SEP cell is \$90.9 per watt or \$11 per cell at 28°C (Reference 6-1, page 6-7). The corresponding cost for the advance-technology cells is \$50 per watt for the high-efficiency cell and \$5 per watt for the low-cost, moderate-efficiency cell. The \$90.9 per watt for the SEP cell translates to \$119 per watt at 80°C because of the reduced output at the elevated temperature. The SCB solar array operates at 80°C. The cost for covering the SEP cell is \$6 per cell, or \$64.7 per watt at 80°C. The SEP cell assembly cost is \$183.7 per watt, or \$17 per cell. The cost for covering the advance-technology cells might be based on either (1) \$6 per cell despite the larger area or (2) a cost proportional to cell area, (\$18.5 per cell). An intermediate value of \$12 per cell was selected. This is an area requiring further study.

The cost of converting the covered cells into a solar array blanket and total power system assembly is estimated to be \$122.5 per watt for the SEP system. The corresponding costs for the advance-technology cells are \$77 per watt for the high-efficiency cell and \$80 per watt for the low-cost cell. Again, the advance technology values are based on an intermediate value between constant cost per cell and cost proportional to cell area. (This area also requires further study). The resulting total costs are \$306 per watt for the SEP system, \$175.5 per watt for the high-efficiency system, and \$125 per watt for the low-cost system.

The total deployable solar array area is  $2,599 \text{ m}^2$  for the SEP system, as previously noted on Figure 6-36. The array area for the advance-technology options is also shown on Table 6-10; it varies inversely with cell efficiency. The SEP cell efficiency is 11.1 percent, bare or 11.4 percent covered; 11.4 percent efficiency is the more commonly quoted number.

The total system cost for the three solar cell options discussed above are presented in Table 6-11. The cost categories are the same as discussed previously in connection with Table 6-5. The principal cost saving for the advance-technology cells occurs in the production item, Solar Array and Support Structure. These differences are the direct result of the factors discussed in connection with Table 6-10. Minor differences also exist in the Drag Propellant Transportation item as a result of differences in solar array area. The Replacement Hardware Transportation item cost is very low (\$2 million) due to the low weight and long life of the regenerative fuel cell system assumed for this example. It is clear that the advance-technology solar cell options are very attractive. Further optimization is required to determine the optimum combination of efficiency and cell cost. In general, it appears that the cost should be lowered to the point where cell efficiency starts dropping rapidly; costs then should be held at that level. It is clear that low-cost solar cell options should be pursued for the SCB application. In addition, further work is required to evaluate means of also reducing cell covering costs and blanket/system assembly costs.

The "Structural/Mechanical/Miscellaneous" DDT&E and production costs are significantly higher in Table 6-11 than for the corresponding items in

Table 6-11

10-YEAR DEPLOYABLE SOLAR ARRAY PROGRAM COSTS  
(MILLIONS OF 1978 DOLLARS)\*

|                                       | Current<br>Cells<br>(11.1%, \$91/W) | High-<br>Efficiency<br>Cells<br>(14%, \$50/W) | Low-Cost<br>Cells<br>(12%, \$5/W) |
|---------------------------------------|-------------------------------------|-----------------------------------------------|-----------------------------------|
| DDT&E                                 |                                     |                                               |                                   |
| Solar array and support structure     | 42                                  | 42                                            | 42                                |
| Energy storage and power conditioning | 40                                  | 40                                            | 40                                |
| Structure/mechanical/miscellaneous    | 55                                  | 55                                            | 55                                |
| Subtotal                              | 137                                 | 137                                           | 137                               |
| Production                            |                                     |                                               |                                   |
| Solar array and support structure     | 84                                  | 48                                            | 34                                |
| Energy storage and power conditioning | 15                                  | 15                                            | 15                                |
| Structure/mechanical/miscellaneous    | 6                                   | 6                                             | 6                                 |
| Subtotal                              | 105                                 | 69                                            | 55                                |
| Operations (10 Years)                 |                                     |                                               |                                   |
| Initial launch transportation         | 40                                  | 40                                            | 40                                |
| Replacement hardware/spares           | 16                                  | 16                                            | 16                                |
| Replacement hardware transportation   | 2                                   | 2                                             | 2                                 |
| Drag propellant transportation        | 40                                  | 33                                            | 38                                |
| Subtotal                              | 98                                  | 91                                            | 96                                |
| Total                                 | 340                                 | 297                                           | 288                               |

NOTE: Shuttle-erected Free-Flyer requires additional \$37 million DDT&E and \$18 million production costs

\*No space fabrication, fuel cell energy storage, solar cells as noted

Table 6-5. This is due to the more complex two-axis gimbaling system, the extension tunnel, and the shirtsleeve environment of the energy storage module compartment for the deployable solar array version.

A summary of the launch weights for the three solar cell options is presented in Table 6-12. The turret/drive assembly includes the structural shell, access provisions, and gimbal rings with the tunnel being a waffled structure with seals. The drive assembly includes the motors, bearings, drive chain, wave generator, and various splines. The module includes the structural shell, berthing port, and allowance for various subsystem interfaces and support. Thermal/meteoroid protection includes a double-layer shield with integral multilayer insulation with appropriate allowances for attachments and external thermal coatings. The radiator includes allowance for the panels, thermal controls, fluids, and thermal coatings. The beam/assembly

Table 6-12  
DEPLOYABLE SOLAR ARRAY WEIGHTS

| Item                                                   | Weight (kg) |           |           |
|--------------------------------------------------------|-------------|-----------|-----------|
|                                                        | 11% Cells   | 14% Cells | 12% Cells |
| Structure                                              | 1,409       | 1,389     | 1,399     |
| • Turret/drive assembly                                | (494)       | (494)     | (494)     |
| • Tunnel                                               | (128)       | (108)     | (118)     |
| • Module                                               | (787)       | (787)     | (787)     |
| Thermal/Meteoroid Protection                           | 294         | 278       | 286       |
| Solar Array                                            | 4,651       | 3,698     | 4,266     |
| • SEP array assemblies                                 | (3,851)     | (3,026)   | (3,530)   |
| • Beam assembly and miscellaneous                      | (800)       | (672)     | (736)     |
| Regenerative Fuel Cells and Support                    | 2,908       | 2,908     | 2,908     |
| Radiator, 215 m <sup>2</sup> (Effective Radiator Area) | 684         | 684       | 684       |
| Power Conditioning and Miscellaneous                   | 1,724       | 1,724     | 1,724     |
| Subtotal                                               | 11,670      | 10,681    | 11,267    |
| Contingency (25%)                                      | 2,918       | 2,670     | 2,817     |
| Total                                                  | 14,588      | 13,351    | 14,084    |

is the 1-meter deep channel with allowance for local support and attachments of the solar panels.

The SEP array assembly weight is based on 20 units at 192.5 kg each. The beam assembly supports the SEP arrays and provides attachment to the gimbal mechanism. The regenerative fuel cell includes fuel cells, electrolysis cells, gas and water tanks, and miscellaneous brackets and plumbing. The power-conditioning item includes inverters, dc voltage regulators, wire, and miscellaneous switch gear and controls. This system is considerably lighter than the power platform solar array (discussed in Section 6.1) because it employs the RFC energy storage system rather than NiCd batteries. The weight differences for the various solar cell technologies is small.

## 6.5 SUMMARY AND CONCLUSIONS

A summary of the candidate energy storage systems is presented in Table 6-13. The launch weight is the weight of the initial installation. The hardware resupply weight is the replacement weight (1 ship-set); the hardware must be replaced at the replacement intervals, either 3.33 years or 5 years depending on the energy storage system. The 10-year weight is the sum of the launch weight, the total hardware resupply weight, and the drag propellant weight. This 10-year weight represents the total weight to orbit chargeable to the electrical power system. The NiCd battery is rated only fair with regard to operations complexity, because of the total weight and number of batteries that must be replaced at 3.33-year intervals. The RFC system is good in this regard because of its longer life and considerably lower weight. The RFC reliability is not quite as good as that of the battery systems because the RFC is less modular in nature, more complex, and has some complicated internal systems. However, the fuel cell and electrolysis cell internal systems are generally maintainable.

All of the battery systems present some concern for safety because of the possibility of cell rupture. The NiCd batteries may release KOH. The  $\text{NiH}_2$  batteries operate at higher pressures (up to 3,448 kPa, 500 psi) with the potential for even greater rupture damage and the release of  $\text{H}_2$  gas. Thus, the  $\text{NiH}_2$  batteries are rated lower than the NiCd batteries. All of the

Table 6-13  
ENERGY STORAGE COMPARISON (BASELINE POWER PLATFORM)

|                                    | NiCd                          | Advanced<br>NiCd              | NiH <sub>2</sub>            | Regenerative<br>Fuel Cell |
|------------------------------------|-------------------------------|-------------------------------|-----------------------------|---------------------------|
| Development/Schedule Risk          | Very low                      | Medium                        | Medium-High                 | Medium                    |
| Launch Weight, kg                  | 34,763                        | 25,868                        | 21,450                      | 16,083                    |
| Hardware Resupply Weight, kg       | 17,470                        | 10,730                        | 7,233                       | 2,087                     |
| Ten Year Weight, kg                | 131,102                       | 104,584                       | 93,836                      | 77,567                    |
| Mission Flexibility <sup>(1)</sup> | Good                          | Good                          | Good                        | Good                      |
| Expected Life, Years               | 3.33                          | 3.33                          | 3.33                        | 5                         |
| Operations Complexity              | Fair                          | Fair +                        | Fair +                      | Good                      |
| Reliability <sup>(2)</sup>         | Good                          | Good                          | Good                        | Fair-good                 |
| Safety                             | Fair +; rupture<br>and shorts | Fair +; rupture<br>and shorts | Fair; rupture<br>and shorts | Fair +; rupture           |
| Cost, \$M                          |                               |                               |                             |                           |
| DDT&E                              | 73                            | 74                            | 75                          | 95                        |
| Ten-Year Total <sup>(3)</sup>      | 349                           | 338                           | 335                         | 317                       |

(1) Alternative applications and power growth

(2) All systems degrade gracefully and are maintainable

(3) Includes beam fabrication module, \$45 million total power system cost



batteries are potentially subject to shorts and arcing during maintenance and replacement operations.

RFC system safety is similar to  $\text{NiH}_2$  system safety except that the RFC system is less subject to inadvertant shorting. The RFC system is rated the same as the NiCd batteries. The RFC system offers a significant savings in total mission cost at the expense of a  $\$20 \times 10^6$  differential in DDT&E costs. In summary, the regenerative fuel cell appears to have a slight edge over the other systems because of lower total mission cost and less operation complexity. The advanced NiCd and  $\text{NiH}_2$  batteries are also deserving of development support as they are extremely attractive. Additional mission definition and system design studies will be beneficial in identifying the optimum energy storage system choice.

The comparisons of Table 6-14 are to some extent "apples and oranges" comparisons in that different energy storage devices, gimbal systems, and solar cell technologies were employed. The advanced/more complex systems of the advanced deployable solar array could be incorporated into the baseline power platform with the same attendant advantages. On the other hand, the power platform offers the opportunity to demonstrate space fabrication and assembly of large solar power sources; this could be very beneficial to the development of SPS technology. The \$349 million cost for the fabrication and assembly platform includes \$45 million for the beam fabrication module. The module cost is included here because the composite beam is being applied for the first time; this is a general purpose device however and its cost perhaps should not be charged to the solar array system. The deployable array assumes the use of the low-cost ( $\eta = 12$  percent) solar cell, and the resulting array area is  $2,404 \text{ m}^2$ , whereas the nominal design is based on current SEP cells and has an area of  $2,599 \text{ m}^2$ .

The following conclusions and recommendations are offered; these are in addition to the SRT recommendations to be discussed subsequently.

- A. Power Platform Program Definition. A power platform program should be defined to further refine the mission requirements and power platform designs. This would provide sharper and continuing

Table 6-14  
COMPARISON OF POWER PLATFORM/ADVANCED DEPLOYABLE  
SOLAR ARRAY SYSTEM CHARACTERISTICS

|                           | Power Platform     | Advanced Deployable<br>Solar Array |
|---------------------------|--------------------|------------------------------------|
| Array                     |                    |                                    |
| Size, m <sup>2</sup>      | 2,407              | 2,404                              |
| $\eta$ Cell (bare)        | 11.1%              | 12%                                |
| Output BOL, kWe           | 214.7              | 230.7                              |
| Output (avg, BOL), kWe    | 100/typical        | 100                                |
| Erection                  | Fab/assy on orbit  | Deploy/assemble on orbit           |
| Orientation               | 1 axis gimbal      | Two-axis gimbal                    |
| Energy Storage            |                    |                                    |
| Type                      | Current tech NiCd  | Regenerative fuel cell             |
| DOD                       | 14.5%              | 33% equivalent                     |
| Life, years               | 3.33               | 5                                  |
| Weight, kg <sup>(1)</sup> | 17,470             | 2,908                              |
| System (10-Year)          |                    |                                    |
| Hardware Weight, kg       | 76,682             | 19,885—                            |
| Shuttle Launches          | 2.7                | 2.1                                |
| Total Cost, \$M           | 349 <sup>(2)</sup> | 288                                |

(1) One ship-set

(2) Includes composite beam fabrication and assembly—\$45 million

focus for the on-going SRT program. Emphasis should be placed on additional evaluation of the low-cost silicon solar cell options, discussed in Section 6.4.3, and on the regenerative fuel cell energy storage system.

- B. Low-Cost Solar Cells. The low-cost solar cell options are very promising. Development emphasis should be placed on the lowest possible cost as long as moderate efficiency (e. g., 10-12 percent) can be maintained. The analysis and development work should include low-cost covers/covering and blanket/solar array assembly.

C. NiCd Battery

1. NWSC Crane test data should be applied cautiously. It may be appropriate to do some work on how to apply these data for system design purposes.
2. The 55 to 66 W-hr/kg energy density suggested by JPL for advanced NiCd batteries appears to be optimistic for manned/maintainable SCB applications in LEO for mid-1985 IOC; 44 to 49 W-hr/kg appears to be more appropriate.

D. NiH<sub>2</sub> Battery

1. This battery has a lot of potential for long-life and high-energy density and its technology should be pursued.
2. LEO life test data are badly needed.
3. A conservative DOD was selected (18.6 percent); it is expected that much higher DODs will be suitable for long life (the system has considerable potential for weight/life improvements over the values selected for SCB design purposes).

E. Regenerative Fuel Cell System

1. The basic Shuttle fuel cell technology has the potential for long life. The same is true for the GE-technology fuel cell and electrolysis cell.
2. A 100-kWe system is more economic, lighter, and less complex with a fewer number of relatively large fuel cells (approximately four times Shuttle size).

We recommend that the following SRT work should be done:

A. Solar Arrays

1. Application analysis and technology development of low-cost solar cells and the related cell covering/blanket assembly technology.
2. Design and development of large solar cell blankets, including continuous blankets with attachments as appropriate for the baseline power platform solar array.
3. Space testing of solar cell blanket samples (e. g., LDEF testing).

B. NiCd Batteries (100-110 AH). Development and demonstration (real time and accelerated) of high energy density, long life batteries.

C.  $\text{NiH}_2$  Batteries (100 AH)

1. Real-time and accelerated life testing for LEO applications.
2. Manned SCB vehicle integration analysis (e. g., maintenance/ replacement, cooling, and charging).

D. Regenerative Fuel Cell System

1. System/vehicle integration analysis (e. g., maintenance/ replacement approaches).
2. Low-cost, long-life fuel cells and electrolysis cells—This work should include (1) preliminary design of a large (approximately four times Shuttle size), maintainable system and (2) analysis and test (accelerated and real-time) to confirm long-life capability.

## Section 7

### ELECTRIC POWER SYSTEMS COMPARISON

The integrated SCB EPS system-level data presented in Sections 4 through 6 are compared in this section. Ten-year-program-level cost data are also presented along with appropriate qualifying guidelines and comments.

These comparative data will allow NASA and ERDA to qualitatively assess the relative worth and cost of the systems studied. It should be noted that the data represent systems in quite different states of technological readiness and at different levels of definition. However, these data should provide reasonable insight into the appropriateness of both supporting research and technology and system-level support in the near future.

#### 7.1 SYSTEM COMPARISONS

##### 7.1.1 System Orbital Deployment/Activation

Results of operational analyses of the orbital deployment/activation of the candidate EPS systems are summarized in Table 7-1. The orbital times range from 7 to 36 shifts; each shift is assumed to be the equivalent of 18 manhours -- 3 EVA astronauts working 6-hour shifts.

The time involved in construction and activation of the baseline solar array system thus represents 576 hours of EVA time. These time estimates were made considering ground rules derived for SCB construction activities; they should, therefore provide a good relative assessment of the systems even if the absolute value cannot be determined.

No significant area was identified in this analysis which required activities and equipment significantly different than those visualized for use in supporting the baseline SCB activities.

Table 7-1  
ORBITAL CONSTRUCTION OF ALTERNATIVE POWER SYSTEMS

| System                                                                                                                                                                                 | Construction Time* (Shifts) | Shuttle Launches | Special Tools                                                                  |
|----------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|-----------------------------|------------------|--------------------------------------------------------------------------------|
| SCB Construction                                                                                                                                                                       |                             |                  |                                                                                |
| • Reactor brayton                                                                                                                                                                      | 7                           | 2                | Assembly support equipment                                                     |
| • Solar brayton                                                                                                                                                                        | 16                          | 2                | -                                                                              |
| • Advanced deployable array                                                                                                                                                            | 18                          | 2                | -                                                                              |
| • Power platform (beam fabrication on orbit)                                                                                                                                           | 32                          | 1                | Composite beam maker                                                           |
| Sortie Mode Construction                                                                                                                                                               |                             |                  |                                                                                |
| • Reactor brayton                                                                                                                                                                      | 8                           | 2                | Docking adaptor and control system                                             |
| • Solar brayton                                                                                                                                                                        | 19                          | 3**              | Docking adaptor, control system, EVA scaffolding; Scaffold-mounted manipulator |
| • Power platform (beam fabrication on orbit)                                                                                                                                           | 35                          | 1                | Control system docking adaptor                                                 |
| <p>*One shift is defined as 6 hours utilizing 3 EVA astronauts, does not include time for contingencies, extensive checkout, etc.</p> <p>**Assumes second unit installed using SCB</p> |                             |                  |                                                                                |

### 7.1.2 System Reliability

The reliability extrapolations discussed in Sections 4 and 5 are summarized in Table 7-2. As can be seen, either the solar brayton system with maintenance or the reactor brayton system without maintenance can be expected to achieve about 0.999 probability of successful 10-year operation with the use of four baseline modules.

If the reactor system electronics, instrumentation, and brayton engine controls are maintainable, the number of baseline modules could be reduced by one (4 to 3). This would result in a program savings of approximately

Table 7-2  
COMPARATIVE 10-YEAR-RELIABILITY ANALYSIS

| Reactor Power Unit |                |                  | Solar Power Unit With Maintenance |               |              |             |
|--------------------|----------------|------------------|-----------------------------------|---------------|--------------|-------------|
| Units*             | <u>120 kW</u>  |                  | Units*                            | <u>120 kW</u> | <u>60 kW</u> |             |
|                    | No Maintenance | With Maintenance |                                   | Reliability   | Units*       | Reliability |
| 1/1                | 0.7686         | 0.8352           |                                   |               | 1/1          | 0.8588      |
| 1/2                | 0.9645         | 0.9838           | 2/2                               | 0.7375        | 1/2          | 0.9884      |
| 1/3                | 0.9952         | 0.9985           | 2/3                               | 0.9768        | 1/3          | 0.9991      |
| 1/4                | 0.9994         |                  | 2/4                               | 0.9982        |              |             |

\*Standby redundancy assumed

\$24.5 million — \$4.5 million for production of one unit and \$20 million for launching that unit.

Since the solar brayton system uses two operating modules to produce full power, an additional mode of half-power operation is available which also would require only three baseline modules. This mode could become primary if early SPS testing does not materialize, or it could merely be available to assure uninterrupted power at the 50-percent level during a module failure. The potential program savings in the event of late mission power growth would be approximately \$22 million — \$2 million for production of one unit and \$20 million for launching that unit.

Solar array reliability data calculated for the Skylab mission is given in Figure 7-1. Modularity and series/parallel configuration of the cells and support circuitry were used to achieve high probability of minimum power degradation for the duration of the mission due to cell/circuitry failure. Extension of these techniques to the much larger (20+ times) solar arrays used in this study should easily allow equivalent results even for the longer-duration mission.

### 7.1.3 EPS Propellant Penalties and Total Launch Requirements

Significant attitude control and orbit-keeping parameters for the SCB are shown as a function of EPS in Table 7-3. Since up to four months 'storage' in space is required of the SCB, the orbital life without orbit-keeping is of particular interest. The solar array systems are both marginal in this respect and may require a Shuttle visit for reboost during this interval or other action to resolve this difficulty.

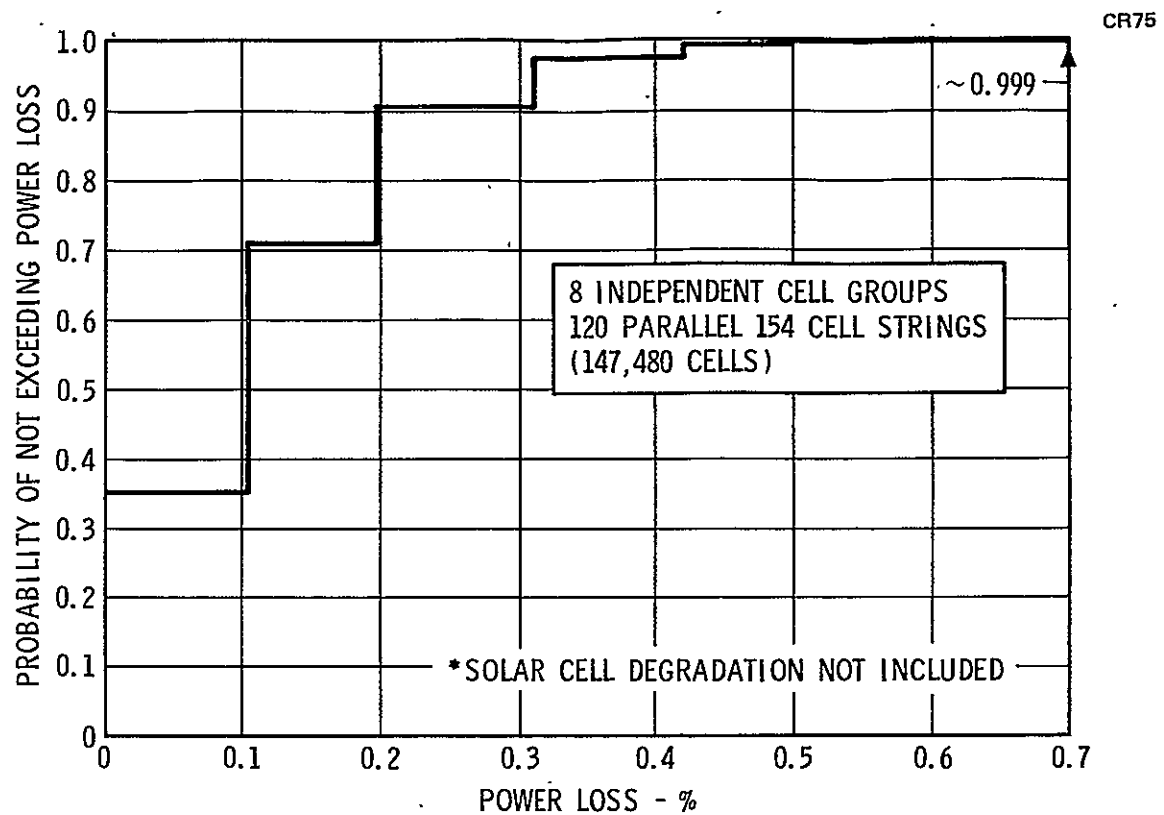


Figure 7-1. End-of-Mission Power Loss Probability for Solar Array System\*

Table 7-3

SCB PROPELLANT USE (NO MISSION HARDWARE)

|                                           | Solar<br>Baseline | Advanced<br>Array | Reactor<br>Brayton | Solar<br>Brayton |
|-------------------------------------------|-------------------|-------------------|--------------------|------------------|
| W/CdA (lb/ft <sup>2</sup> )               | 2.9               | 2.6               | 26.1               | 8.75             |
| Orbit-Keeping Interval<br>(days)          | 5.8               | 5.1               | 51                 | 17               |
| Orbit Life (weeks)                        | 10                | 9.2               | 92                 | 33               |
| Propellant-Drag (kg/month)                | 476               | 431               | 69                 | 150              |
| CMGs Used                                 | No                | No/Yes*           | No                 | No/Yes*          |
| Propellant-Attitude Control<br>(kg/month) | 186               | 39/19**           | 108                | 215/25**         |
| Total Propellant Use<br>(kg/month)        | 662               | 470/450**         | 177                | 365/175**        |

\*Baseline

\*\*No CMG/CMG

The reactor and solar brayton systems both have large margins for meeting this requirement.

The drag makeup (orbit-keeping) propellant use for each system is most significant regarding 10-year support costs because the use of attitude



control propellant is primarily due to the basic SCB (i.e., it would be 80 percent of the values shown even without an EPS).

A summary of 10-year launch requirements established by the alternative EPSs is given in Table 7-4. While the baseline solar array requires by

Table 7-4  
LAUNCH REQUIREMENTS COMPARISON FOR A 10-YEAR PROGRAM

|                         | Solar<br>Baseline          | Advanced<br>Array         | Solar Brayton             | Reactor Brayton           |
|-------------------------|----------------------------|---------------------------|---------------------------|---------------------------|
| Initial System (kg)     | 34,763                     | 16,977                    | 26,125                    | 36,640                    |
| Replacement System (kg) | ---                        | ---                       | 23,625                    | 34,250                    |
| Spares (kg)             | 41,919                     | 3,757                     | 2,310                     | 3,370                     |
| Propellant (kg)         | 54,422                     | 49,239                    | 17,959                    | 8,163                     |
| Total Weight to Orbit   | 131,104 kg<br>(289,084 lb) | 69,973 kg<br>(154,290 lb) | 70,019 kg<br>(154,392 lb) | 82,423 kg<br>(181,743 lb) |
| Total Launches          | 1+0. 2+1. 5+1. 9<br>4. 6   | 2. 0+0. 2+1. 7<br>3. 9    | 4+0. 1+0. 6<br>4. 7       | 4+0. 1+0. 2<br>4. 3       |

far the greatest weight to be launched to orbit, due primarily to battery replacement and propellant use, the variation in launches is not large. This is due to the fact that both the reactor and solar brayton systems have volume-limited modules which make inefficient use of the Shuttle capability. Thus there is no great advantage from a transportation cost/support operations standpoint.

#### 7.1.4 Alternative Mission Capability

A brief assessment of the compatibility of the various EPSs with orbits other than the baseline of 28.5 degrees and 400 km is presented in Figure 7-2.

The curves indicate the Orbiter payload capability as a function of altitude; the arrows on the left of the figure represent the EPS launch weights (including any penalties added due to orbit change). It is possible to launch all systems to 45 degrees and 550 km; the solar brayton system can be launched to 55 degrees and 600+ km. All systems can also be launched to 55 degrees and 450 km.

Although not indicated on the figure, none of the systems as presently configured can be launched into a 90-degree orbit without either repackaging into two or more launches or use of an orbital 'tug'.

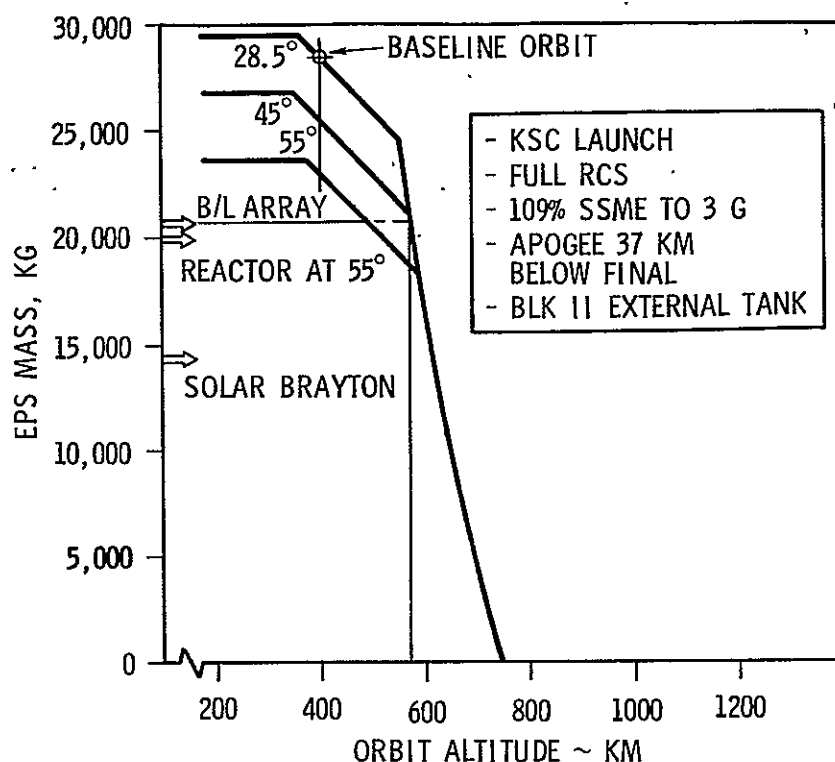


Figure 7-2. EPS Alternate Mission Capability

#### 7.1.5 EPS Safety

Current definitions of EPS candidates are not adequate for conventional fault tree and failure modes and effects analyses to be undertaken. However, it is possible to identify major tasks and events which could impact crew and mission safety. Table 7-5 indicates some of these major safety issues.

As indicated, adequate crew workaids and safeguards must be provided to assure crew effectiveness and safety during EPS installation and support operations. This appears to be a straightforward design problem.

In addition, careful system and mission detail design and planning, and thorough safety analyses at each step of system development, will be required to attain acceptable risk. Obviously the reactor brayton system with the most potential problems will take the most concerted effort, with the attendant costs, for resolution.

Table 7-5  
EPS SAFETY ISSUES

|                                                                                                                                | Solar Array(s)                          | Solar Brayton                          | Reactor Brayton                               |
|--------------------------------------------------------------------------------------------------------------------------------|-----------------------------------------|----------------------------------------|-----------------------------------------------|
| Crew                                                                                                                           | EVA for installation                    | EVA for installation                   | EVA for installation                          |
|                                                                                                                                | EVA for battery replacement/maintenance | EVA for system replacement/maintenance | EVA for system replacement/maintenance        |
|                                                                                                                                |                                         |                                        | Added radiation dose during maintenance       |
| Adequate crew work aids/safeguards must be provided                                                                            |                                         |                                        |                                               |
| System/<br>Mission                                                                                                             | Total power loss                        | Total power loss                       | Total power loss                              |
|                                                                                                                                |                                         | Accidental release of lithium fluoride | Accidental release of mercury from heat pipes |
|                                                                                                                                |                                         |                                        | Accidental release of fuel                    |
|                                                                                                                                |                                         |                                        | Failure/malfunction of disposal system        |
|                                                                                                                                |                                         |                                        | Loss or runaway of reactor                    |
| Careful system/mission detail design/planning coupled with thorough safety analysis should allow attainment of acceptable risk |                                         |                                        |                                               |

#### 7.1.6 EPS Technical Risk

The expected technical risk associated with use of the EPS alternatives is summarized in Table 7-6. The data of the table are based on the technical risk and SRT information provided in Sections 4 through 6 of this report.

The solar array systems draw upon proven technology — from early satellites through Skylab — and are considered suitable for the start of detail design. Where SRT items have been identified (e.g., fuel cell energy storage), alternative approaches are available. System readiness therefore does not

have to depend upon development testing of these improved elements — rather they add to the attractiveness of the solar array system if time and cost permit their use.

The solar brayton system concentrator and receiver, on the other hand, require further detail definition and test prior to commitment to detailed design. Certainly, adequate preliminary work has been done to indicate a reasonable probability of success in these areas. The brayton engine requires scaling up in size from the LeRC B engine, but experience with open-cycle brayton jet engines provides reasonable assurance of success.

Table 7-6  
TECHNICAL RISK

| Solar Array                                      | Solar Brayton                                      | Reactor Brayton                                                     |
|--------------------------------------------------|----------------------------------------------------|---------------------------------------------------------------------|
| Array/energy storage control system-Low          | Concentrator/receiver/gimbal/control system-Medium | Reactor/control system-High                                         |
|                                                  | Brayton engine system-Low                          | Brayton engine system-Medium                                        |
| Radiator system pumped loop-Low                  | Radiator system pumped loop-Low                    | Radiator system pumped loop-Low<br>Radiator system heat pipe-Medium |
| Energy conversion, control, and distribution-Low | Energy conversion, control and distribution-Low    | Energy conversion, control and distribution-Low                     |

The new heat pipe reactor concept appears to provide the probability of high reactor reliability and safety. It does however depend on extrapolations in temperature/materials compatibility for much longer periods of time than we have test data for today. Furthermore, much of the required testing must be done 'in core' which is both time-consuming and expensive. The brayton engine associated with the reactor system requires the use of refractory materials; this is in contrast with the superalloys used for the solar brayton. Development of such an engine — and its testing — will be both difficult and expensive (much manufacturing and all testing must be done in an oxygen-

free environment). Therefore, the reactor brayton engine must be classified as higher risk than the solar brayton engine. The current heat pipe radiator concept is actually a hybrid using pumped headers to feed the radiator heat pipes. This fact, coupled with the quite complex and presently poorly defined heat pipe interfaces, suggests that the added cost and complexity of the proposed system relative to a redundant pumped loop radiator is not warranted.

In summary, from a relative technical risk standpoint it appears that solar arrays are the obvious choice, with solar brayton being medium risk and reactor brayton high risk.

#### 7.1.7 Growth Capability of EPS Alternatives

For solar array systems, (1) the deployable system is limited to about 50 to 60-kW per launch, (2) the baseline (space-fabricated) system is limited to about 120 kW per launch, (3) larger panels significantly increase orbit-keeping penalties and operational complexity, and (4) unit costs are high. Cheaper solar cells and advanced energy storage systems will drastically improve the competitive position of these systems.

For the solar brayton system, (1) the concentrator size limits the power level to about 70 to 75-kW per launch, (2) weight growth does not appear critical, and (3) cycle efficiency gains can permit significant power level increases.

For the reactor brayton system, (1) deployable radiators will be required to allow power levels above 120 to 150 kW per launch, but (2) an adequate weight margin is available for added shielding, and for shield growth if required.

Thus all systems, as presently conceived, are close to the maximum power level available without requiring additional launches and more complex orbital assemblies. The solar array systems will continue to have higher unit costs and drag makeup propellant penalties, and will constrain operations in the largest volume of space around the vehicle.

For much larger power plants (200 to 500 kWe average power) the reactor brayton system will cause the least interference with operations and construction and will have the most reasonable specific weights.

## 7.2 COST AND SCHEDULE

The following general guidelines were utilized in development of comparative costs and schedules for the EPS alternatives.

- A. Use LASL and LeRC costs and schedules for comparisons.  
Use historical comparison for risk assessment since designs do not permit independent cost analysis.
- B. Develop supporting hardware, integration, assembly, maintenance, and launch costs consistent with SSSAS results.
- C. Assume SCB baseline program for schedule comparisons (mid '85 IOC).
- D. Costs are in fiscal 1978/dollars.

More specific costing ground rules will be found in Appendix 5.

Since the reactor brayton concept was not detailed enough to allow MDAC to prepare an independent cost assessment, costs developed by MDAC and

Table 7-7  
HISTORICAL NONRECURRING COSTS

|                               | Phase B<br>Reactor Brayton |       | SSSAS |
|-------------------------------|----------------------------|-------|-------|
|                               | NAR                        | MDAC  | LASL  |
| Reactor                       | 53.1                       | 54.6  | ---   |
| PCS                           | 89.7                       | 50.3  | 37.3  |
| TCD                           | 30.7                       | 13.7  | ---   |
| Disposal packaging            | 0                          | 19.8  | ---   |
| Structure                     | 128.0                      | 19.8  | ---   |
| Subsystem                     | 61.3                       | 4.4   | ---   |
| GSE                           | 50.0                       | 12.9  | ---   |
| Facilities                    | 30.0                       | 10.1  | 11.0  |
| System test operator training | 84.6                       | 44.8  | ---   |
| System Support                | 27.3                       | 0.8   | ---   |
| Totals:                       |                            |       |       |
| (Millions of 1970 dollars)    | 554.7                      | 231.2 | ---   |
| (Millions of 1978 dollars)    | 910.3                      | 379.4 | 179.9 |

Table 7-8  
HISTORICAL RECURRING COSTS

|                            | Phase B<br>Reactor Brayton |      | SSSAS |
|----------------------------|----------------------------|------|-------|
|                            | NAR                        | MDAC | LASL  |
| Reactor                    | 13.6                       | 11.5 | 2.7   |
| PCS                        | 29.0                       | 13.1 | 1.56  |
| Contingencies              | ---                        | ---  | 0.5   |
| <u>Totals:</u>             |                            |      |       |
| (Millions of 1970 dollars) | 42.6                       | 24.6 | ---   |
| (Millions of 1978 dollars) | 74.2                       | 42.9 | 4.76  |

NAR during the Phase B Space Station studies\* conducted in 1969 and 1970 were reviewed. These costs are compared with the current LASL estimates in Tables 7-7 and 7-8. As can be noted the non-recurring (DDT & E) costs, when escalated to 1978 dollars, ranged from 2 to 5 times higher than the current LASL estimate and the recurring costs from 9 to 15 times higher.

The breakdown of LASL cost data makes a more detailed comparison impossible; however, detailed examination of the various system cost elements for the new reactor concept failed to identify areas where significant cost savings could be made. Rather it appears that inherent complexities in manufacture and assembly of the heat-pipe reactor coupled with the need for refractory materials in both the brayton engine and reactor preclude optimistic cost projections until technology development demonstrates low system costs. For study purposes, a factor of 2 was used for DDT&E and production costs.

While the detailed definition of the solar brayton system was somewhat better than the definition of the reactor brayton system, it too was suspect; however, similar historical data were not available for comparison. As previously noted, the design uncertainty is mainly in the concentrator and receiver; since these are far less complex than a heat pipe reactor, a 1.5 factor for DDT&E and production costs was used to define a probable upper bound for the solar brayton program costs.

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\*These were ~\$200,000 studies of a reactor brayton EPS and involved MDAC/NAR, Atomics International, and AiResearch.

In the case of the solar array systems, although the technology is ready, a detailed definition has not been accomplished. The baseline solar array has a significant cost risk in development of orbital fabrication and assembly equipment and techniques, while the advanced deployable array has significant cost risks in ensuring the availability of advanced solar cells and energy storage system. It was judged that 15 to 25 percent would be reasonable for these near-ready systems, and a factor of 1.25 for DDT&E and production costs was used in defining the probable upper cost bound for these systems.

For solar photovoltaic systems, the costs of the baseline power platform and the advanced deployable array (left and right hand columns) are presented in Figure 7-3. As can be seen, the two power system costs are nearly equal. Lower production and operating costs for the advanced deployable array are offset by increased DDT&E costs incurred early in the program. These DDT&E costs are for the more complex deployment mechanisms, 2-axis gimbaling, and regenerative fuel cell energy storage developments.

The center two columns of Figure 7-3 give cost projections assuming application of either low cost solar cells or regenerative fuel cells to the baseline power platform. As can be seen, either technology advance provides a reduction in total power system costs of about 10 percent. Further, the selection of low-cost solar cells permits the lowest production and DDT&E costs. It should also be noted that the long-life projections for NiCd batteries (Reference 6-3) by NASA Goddard personnel could further reduce the costs of this power platform option. Additional cost analysis in this area should be beneficial.

Similar comparative data for the solar and reactor brayton systems are given in Figure 7-4. The DDT&E and production costs, except for the small integration package delta, are data supplied by LeRC and LASL. Operations costs are nearly equivalent, differing only because of the higher drag make-up propellant required by the solar brayton system.

The total cost for the solar brayton system using these data is \$232 million. A probable upper-bound cost of \$296 million is also shown utilizing a 50 percent uncertainty on DDT&E and production costs to allow for the



|                                           | SOLAR PHOTOVOLTAIC                       |                                            |                                    |                                |
|-------------------------------------------|------------------------------------------|--------------------------------------------|------------------------------------|--------------------------------|
|                                           | POWER PLATFORM                           |                                            | SEPS SOLAR CELLS<br>AND FUEL CELLS | ADVANCED<br>DEPLOYABLE ARRAY** |
|                                           | SEPS SOLAR CELLS *<br>AND NiCd BATTERIES | LOW-COST SOLAR CELLS<br>AND NiCd BATTERIES |                                    |                                |
| <b>DDT&amp;E</b>                          |                                          |                                            |                                    |                                |
| SOLAR ARRAY AND SUPPORT STRUCTURE         | 36                                       | 50                                         | 36                                 | 42                             |
| ENERGY STORAGE AND POWER CONDITIONING     | 18                                       | 18                                         | 40                                 | 40                             |
| STRUCTURE/MECH/MISC                       | 19                                       | 19                                         | 19                                 | 55                             |
| SUBTOTAL                                  | 73                                       | 85                                         | 95                                 | 137                            |
| <b>PRODUCTION</b>                         |                                          |                                            |                                    |                                |
| SOLAR ARRAY AND SUPPORT STRUCTURE         | 70                                       | 27                                         | 70                                 | 34                             |
| ENERGY STORAGE AND POWER CONDITIONING     | 32                                       | 32                                         | 20                                 | 20                             |
| STRUCTURE/MECH/MISC                       | 2                                        | 2                                          | 2                                  | 6                              |
| SUBTOTAL                                  | 104                                      | 61                                         | 92                                 | 60                             |
| <b>SUBTOTAL, DDT&amp;E AND PRODUCTION</b> | <b>177</b>                               | <b>148</b>                                 | <b>187</b>                         | <b>197</b>                     |
| <b>OPERATIONS (10 YEARS)</b>              |                                          |                                            |                                    |                                |
| INITIAL LAUNCH (\$20M/FLIGHT)             | 24                                       | 24                                         | 24                                 | 40                             |
| SUPPORT LAUNCHES                          | 30                                       | 30                                         | 2                                  | 2                              |
| SPARES AND REPLACEMENT HARDWARE           | 33                                       | 33                                         | 16                                 | 16                             |
| RCS PROPELLANT COST                       | 40                                       | 40                                         | 40                                 | 38                             |
| SUBTOTAL                                  | 127                                      | 127                                        | 82                                 | 96                             |
| <b>TOTAL</b>                              | <b>304</b>                               | <b>275</b>                                 | <b>269</b>                         | <b>293</b>                     |
| <b>PROBABLE UPPER BOUND</b>               | <b>348</b>                               | <b>312</b>                                 | <b>316</b>                         | <b>342</b>                     |

NOTE: SHUTTLE-ERECTED FREE-FLIGHT POWER PLATFORM REQUIRES ADDITIONAL \$37 MILLION DDT&E AND \$18 MILLION PRODUCTION COST  
COMPOSITE BEAM FABRICATION AND ASSEMBLY COST DELETED FROM POWER PLATFORM COSTS: EQUIPMENT  
USED ON OTHER CONSTRUCTION.

\*BASELINE

\*\*NO SPACE FABRICATION — FUEL CELL ENERGY STORAGE — 12% — \$5/W SOLAR CELLS

Figure 7-3. Ten-Year Solar Array Program Costs (Millions of 1978 Dollars)

|                                 | SOLAR<br>BRAYTON | REACTOR<br>BRAYTON |
|---------------------------------|------------------|--------------------|
| DDT&E                           |                  |                    |
| POWER MODULE                    | 103              | 179                |
| INTEGRATION PACKAGE             | 14               | 21                 |
| SUBTOTAL                        | 117              | 200                |
| PRODUCTION                      |                  |                    |
| POWER MODULES (4)               | 8                | 18                 |
| INTEGRATION PACKAGE             | 2                | 3                  |
| SUBTOTAL                        | 10               | 21                 |
| SUBTOTAL DDT & E AND PRODUCTION | 127              | 221                |
| OPERATIONS (10 YEARS)           |                  |                    |
| INITIAL LAUNCH (2)              | 40               | 40                 |
| SUPPORT LAUNCHES (2.1)          | 42               | 42                 |
| SPARES AND REPLACEMENT HARDWARE | 5                | 5                  |
| RCS PROPELLANT COST             | 18               | 9                  |
| SUBTOTAL                        | 105              | 96                 |
| TOTAL                           | 232              | 317                |
| PROBABLE UPPER BOUND*           | 296              | 538                |

\*BASED ON HISTORICAL AND RISK ASSESSMENT

Figure 7-4. Ten-Year Solar and Reactor Brayton System Program Costs (Millions of 1978 Dollars)

limited system definition and the technical risk in the concentrator receiver area.

Total cost for the reactor brayton system using the noted data is \$317 million. A probable upper bound cost of \$538 million is also shown, utilizing a 100 percent uncertainty on DDT&E and production costs to allow for an even more limited system definition and the technical risks in the heat pipe reactor and subsystem elements.

Figure 7-5 compares the energy cost in dollars per kilowatt-hour for these four systems. These costs were calculated using only the recurring costs derived above; thus, they are indicative of the expected cost of using one of these systems on a second or subsequent program without significant modification. The cross-hatched area is defined by the calculated recurring cost at the lower end and the application of the same uncertainty factor utilized above to the upper end.

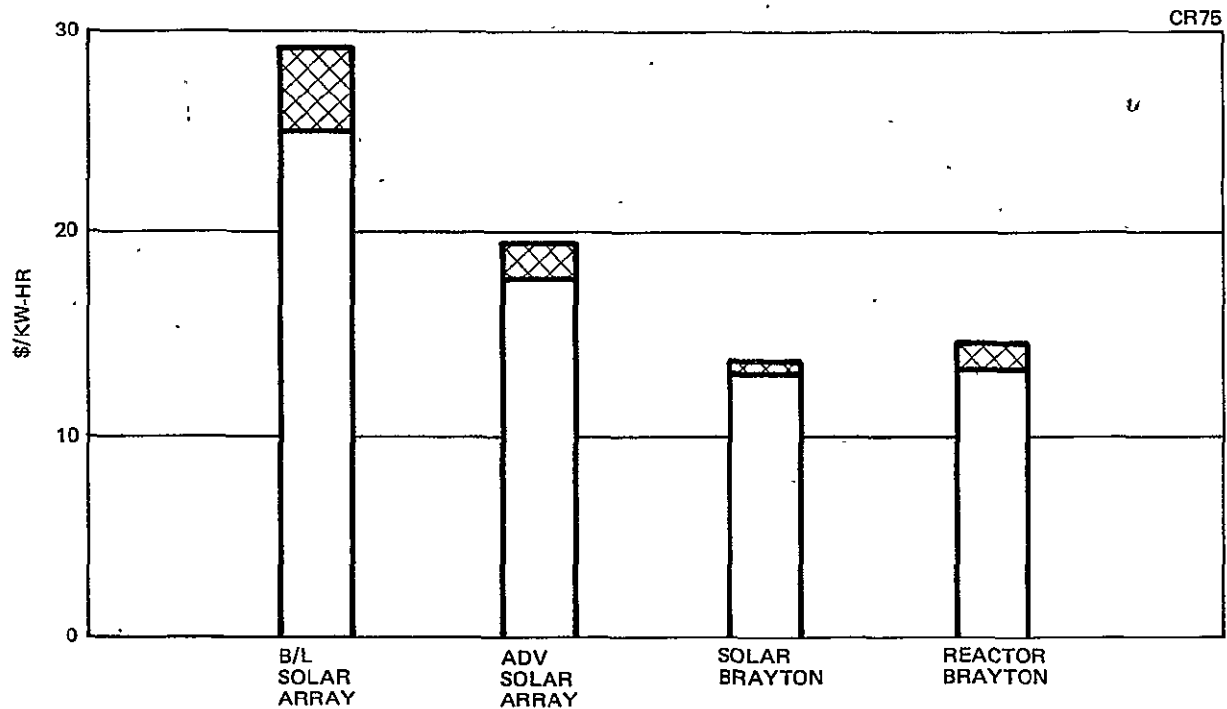


Figure 7-5. Energy Costs for 10-Year Program (DDT&E Excluded)

It appears that successful development of either alternative power system could result in the lowest energy cost for multi-program usage of a power system, based on NASA LeRC and ERDA-LASL recurring cost estimates. However, these estimates need further analysis to verify their accuracy.

A gross comparison of development schedules for the four alternative systems is given in Figure 7-6.

The top line of this figure gives the overall SCB development schedule visualized at the time this study was begun; it shows a mid-1980 start for Phase C/D and a mid-1985 IOC.

The technology status for the solar array system is adequate to begin detailed design at the start of Phase C/D and meet the mid-1985 IOC, as indicated by the second line.

Schedules for the reactor and solar brayton systems were provided by LeRC and LASL, and were also aimed at meeting mid-1985 IOC. However,

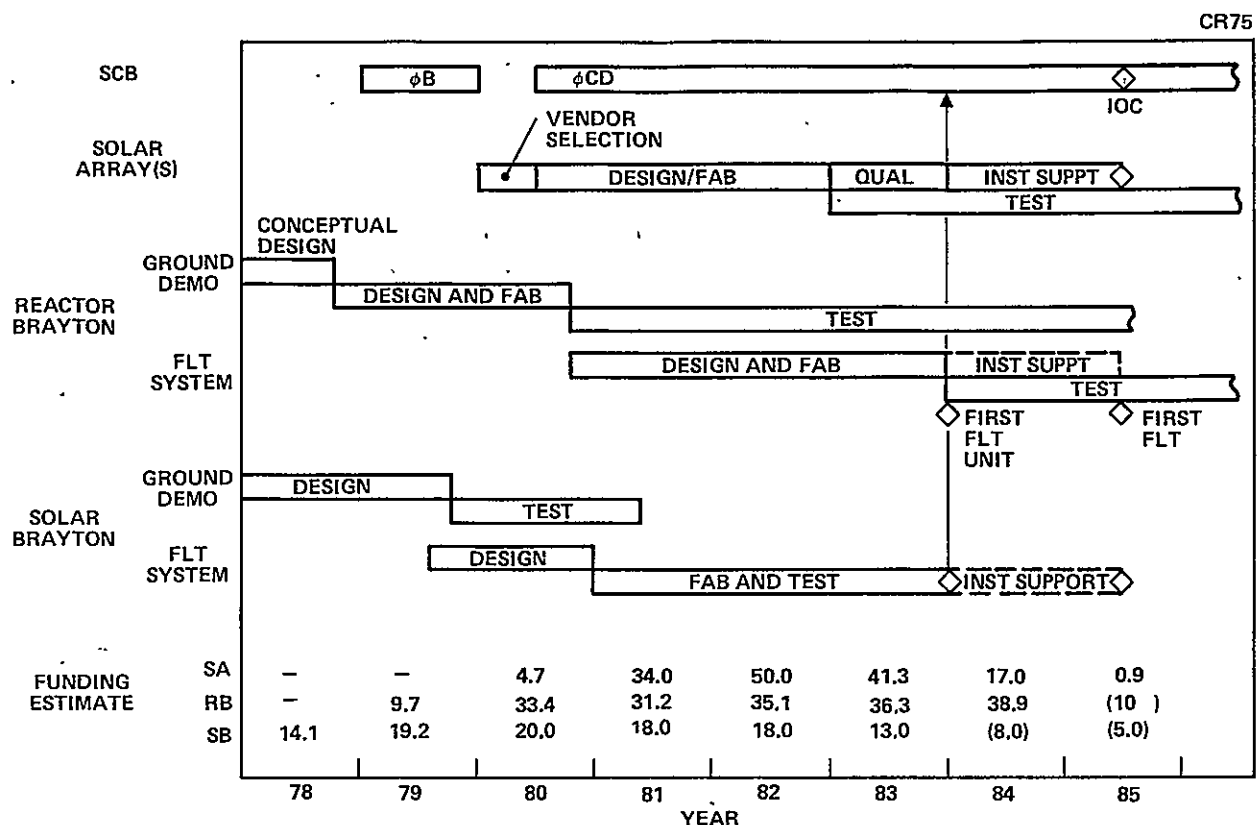


Figure 7-6. EPS Development Programs

both groups believed it necessary to start design prior to the start of Phase C/D of the SCB; we concur that this is necessary, but point out that it is totally unrealistic to expect funding support from the SCB program at these earlier dates. Thus, unless alternate funding of significant magnitude is available, these systems cannot meet a mid-1985 IOC. Further, both schedules show the start of flight system design before significant ground demonstration system testing occurs. We believe this is at the very least apt to be quite expensive and from an SCB program manager's viewpoint, unacceptable. Most likely, at least a year of ground demonstration testing should be included before the flight system design is begun, which would further lengthen the schedules shown.

Since the reality of a mid-1985 IOC for an SCB is in serious question at this time, elimination of either the reactor or solar brayton systems for this reason is not recommended. However, more realistic EPS schedules should be defined for comparison with the still evolving needs of the Space Station and Space Construction Base.

Power system comparisons for the study criteria provided are reported in Section 1 of this report.

Section 8  
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Appendix A

SCB-EPS POWER SOURCE REQUIREMENTS  
MODIFIED JULY 1977

The following requirements have been derived from JSC-11867, SSSA Design Guidelines and Criteria Document, dated January 1977.

- 2.1 The EPS shall provide 50 kWe average power, 115/230 vac, 3 $\phi$ , 400 ~ to The SCB Power Management System (12.01, modified).
- 2.1.1 The EPS shall provide 120% of the average power for peak loads for one hour, three times per day. Peak power of ~450 kW for 15 minutes per orbit shall also be provided (derived).
- 2.1.2 Deleted
- 2.1.3 The EPS shall accommodate the capability for up to 100% growth including both configuration compatibility and electrical compatibility (12.04, modified).
- 2.1.4 The EPS shall have a maintained lifetime of not less than 10 years; however, elements of the EPS may be replaced in total or in modular form for maintenance or for growth. As a design goal, during this maintenance or uprating period the required electrical power levels will be sustained without interruption; as a minimum, 50% power level shall be maintained (12.05, modified).
- 2.1.5 For photovoltaic EPS, energy storage for eclipsed (dark side) power supply shall be by batteries, regenerative fuel cells, or energy-momentum wheels, where energy is restored by power from the solar EPS during light side operations (12.06).
- 2.1.6 Non-photovoltaic power systems may provide for peak power requirements in any of the following modes: (1) using an energy storage system, (2) operating in a load-following mode, or (3) operating continuously at peak power and dumping the unused power, as necessary (2.1.5, above).
- 2.1.7 The electrical power subsystem shall provide both dc and ac service to users at TBD voltage levels with a TBD power split (13.03).
- 2.2 The EPS shall be transportable to and from LEO via the Shuttle-Space Transportation System (1.01).

- 2.2.1 The EPS allowable weight shall be 80% of the Shuttle Launch capability (1.08 and 3.07).
- 2.2.2 SCB IOC shall be mid-1985 (1.02).
- 2.2.3 The LEO parameters for the SCB shall be within a 28.5° to 90° inclination range and a 370km (200 nmi) to 650km (350 nmi) altitude range (1.03, modified).
- 2.2.4 The EPS development approach will provide for reducing the number and cost of test articles and major tests and will provide for utilization of the Shuttle for on-orbit testing, as necessary (1.05).
- 2.2.5 Total cost of the SCB program is a primary consideration. Primary emphasis is on minimum cost including recurring costs through the initial SCB operational period, FY 1985 to 1987 (1.07).
- 2.2.6 The maximum external dimensions of the EPS module shall be 4.42m (14.5 ft) in diameter and 16.2m (53 ft) with planned EVA, or 18.25m (60 ft) in length without planned EVA. Mechanisms that are external but attached to the module, such as handling rings, attachments for deployment, docking mechanisms, storage fittings, thrusters, etc., shall be contained during launch within a dynamic envelope of 4.6m (15 ft) diameter and 18.25m (60 ft) length (1.09, modified).
- 2.2.7 The EPS design for launch via the STS shall meet the payload requirements (e.g., center of gravity and structural loading requirements) as established in JSC 07700, Volume XIV, Rev D, Change No. 19, dated 2 December 1976, both for launch and return landing (1.10).
- 2.2.8 Deleted
- 2.2.9 The on-orbit EPS shall be capable of being placed into a standby unmanned mode and be reactivated after a period of up to 4 months (2.01).
- 2.2.10 There is no requirement that the SCB provide an artificial gravity environment (3.02).
- 2.2.11 The EPS shall provide for checkout, monitoring, warning, and fault isolation to a level consistent with safety and the in-orbit maintenance and repair approach selected (4.04).



- 2.2.12 The EPS shall be designed to minimize risk of loss of SCB modules, injury to the crew, or damage to the Orbiter and other vehicles (fail operational/fail safe) (4.07).
- 2.2.13 If the first module to be orbited is a power module, it shall provide the following communications: telemetry, metric tracking, and when manned, duplex voice links. Subsequent attached modules can rely on the first module for these services (6.05, modified).
- 2.2.14 The initial SCB subsystems shall provide an interface capability with the Orbiter subsystems during Orbiter-tended operations (6.08).
- 2.2.15 Handholds, handrails, and restraint attach points must be provided along all EVA/IVA routes and at each EVA hatch (7.06).
- 2.2.16 A minimum of four feet clear length is required from the EVA hatch located in the Orbiter cargo bay for planned crew EVA (if EVA from Orbiter is required with EPS payload) (7.12).
- 2.2.17 The astronaut radiation exposure allocable to nuclear power systems shall be 12.5 rem/3 mo. This radiation dose may be allowed for two consecutive quarters within a 6 month time period provided that total astronaut radiation exposures do not exceed the quarterly limits established respectively for skin, eyes, and blood-forming organs. This allocation assumes that exposure to solar flare radiation will be limited to  $\leq 10$  rem per quarter by astronaut use of a biowell during all solar flare activity (derived from 11.01). Note: This radiation dose allocation also assumes no sharing of radiation limits with microwave radiation.
- 2.2.18 Safety factors used for structural design shall be consistent with those currently used for manned operations.

Strength

Ultimate - A 1.5 safety factor shall be applied to the ultimate strength for unpressurized structure and 2.0 for pressurized structure.

Yield - A 1.1 safety factor shall be applied to the yield strength for unpressurized structure and 1.5 for pressurized structure.

#### Fail-Safe Structure

- The structure shall be designed so that a credible failure mode in the structure shall not result in a catastrophic failure (9.05).
- 2.2.19 Meteoroid protection shall be provided by the space construction base design consistent with the meteoroid flux given in TM X-53865, second edition, dated August 1970 (9.06).
- 2.2.20 Dynamic isolation is required for rotating machinery (9.08).
- 2.3 The EPS design shall provide a data interface to the SCB data management system for purpose of subsystem status monitoring, trend analysis, out-of-limit caution and warning alarms, fault detection, and fault isolation (15.0).
- 2.4 The EPS design shall be compatible with the following SCB safety and reliability requirements.
- 2.4.1 The EPS shall provide the capability for performing critical functions at a nominal level with any single component failed or with any portion of a subsystem inactive for maintenance (A 2.1, modified).
- 2.4.2 The EPS shall provide the capability to perform critical functions at a reduced level with any credible combination of two component failures, or with any credible combination of a portion of a subsystem inactive for maintenance and failure of a component in the remaining portion of the subsystem (A 2.2, modified).
- 2.4.3 Capability shall be provided for performing critical functions at an emergency level until the affected function can be restored or the crew returned to earth:
- o With any one module inactivated or isolated and vacated due to a malfunction or accident
  - o With any credible combination of a subsystem inactivated as a result of an accident and a portion of a redundant or back-up system inoperative (A 2.3).
- 2.4.4 For those malfunctions which may result in time-critical emergencies, provision shall be made for the automatic switching to a safe mode of operation and for caution and warning of crew members (A 2.4).

- 2.4.5 Capability shall be provided for extinguishing any fire in the most severe oxidizing environment prior to failure of primary structural elements (A 2.5).
- 2.4.6 All continuous nonmetallic materials shall be self-extinguishing in the most severe oxidizing environment to which they will be exposed (A 2.6).
- 2.4.7 Deleted
- 2.4.8 The capability shall be provided for crew survival for at least 180 hours in LEO and TBD hours in GEO to permit restoration of operations or rescue of the crew by emergency Shuttle following any credible combination of component failures, or any credible combination of component failures and portions of a subsystem inactive for maintenance, or any credible accident (e.g., loss of any pressure isolatable volume), and any single component failure (B 2.2).
- 2.4.9 During buildup/premanning the SCB shall be capable of being manned (shirtsleeve or IVA) for performance of maintenance and station assembly tasks following any one component failure (B 2.3).
- 2.4.10 Subsystem or component failures shall not propagate sequentially. Equipment shall be designed to be fail operational/fail safe (B 2.4).
- 2.4.11 All critical life limited components and subsystems shall be designed to allow ground and on-orbit inspection (B 2.5).
- 2.4.12 Equipment or material sensitive to contamination shall be handled in a controlled environment. Fluids and materials shall be compatible with the combined environment in which they are employed (B 2.6).
- 2.4.13 Loss of redundancy for critical functions shall be detectable automatically by the information subsystem and the crew (B 2.7).
- 2.4.14 Redundant paths, such as fluid lines, electrical wiring, and connectors shall be located so that an event which damages one path is not likely to damage the other (B 2.8).
- 2.4.15 Conservative factors of safety shall be provided where critical single failure point modes of operation cannot be eliminated (pressure vessels, pressure lines, valves, etc.) (B 2.10).

2.4.16 All of the subsystems that incorporate an automated fail/operational capability will be designed to provide crew notification and data management system cognizance of component malfunction until the anomaly has been corrected (B 2.10).

## Appendix B

### SOLAR BRAYTON SPACE POWER SYSTEM

#### Introduction

The Lewis Research Center has been engaged in a program to evaluate and demonstrate technology for closed Brayton cycle power conversion systems for space since the early 1960's. With the contracted support of the aerospace industry, a 2 to 15 kWe system has been developed and extensively tested. System parametric studies, a system design and some component development were conducted for multihundred kWe reactor-powered Brayton system. The components for 0.5 to 2.0 kWe Brayton system are presently being fabricated for evaluation and demonstration.

Using this technology, a Brayton system can be developed to meet the thermal and electrical requirements of the Space Construction Base (SCB). The flexibility of this system offers the SCB designer a wide choice in a trade-off of redundancy, reliability, performance and weight. Single or multiple modules can be used up to about 50 kWe. Above that power level multiple modules may be required. Single module output is limited by the practical size of a collector that can be transported and deployed or erected in space.

A representative system with an 11 kWe output is described for discussion purposes. The converter for this system has accumulated in excess of 26,000 hours operation; the full-scale radiator has been tested in vacuum and the receiver and collector have been fabricated but not tested.

#### Description

A solar Brayton engine can be produced to satisfy a variety of onboard electric power and thermal energy requirements of an orbiting spacecraft for both housekeeping and experimental activities. Electric power to about 50 kWe can be furnished by single or multiple modules. The use of multiple modules may be required for larger power needs.

Lewis Research Center has designed a solar Brayton system and fabricated the components for an 11 kWe module. A description of this module is presented herein as being typical of a module that could supply the power needs of the Space Construction Base. This system was designed and fabricated based on the available technology of the mid '60's.

The module consists of a solar collector, heat receiver, power conversion loop, heat rejection loop and an electrical control package. A schematic of the complete module is presented in figure 1 (Ref. 1) which depicts the major components and a consistent set of system parameters for the 11 kWe output.

The power conversion loop (converter), using a mixture of helium and xenon gas as the working fluid, contains the rotating unit (turbine-

alternator-compressor), the recuperator and the gas portion of the gas-to-liquid waste heat exchanger. The heat rejection loop contains the liquid portion of the waste heat exchanger and the radiator to reject the waste heat to space. The recuperator and the waste heat exchanger have been fabricated in a single package called the Brayton Heat Exchanger Unit (BHXU).

Each of the major components of the whole system (including converter components) is described below.

Solar Mirror - a lightweight, deployable parabolic mirror is postulated for the solar heat source. The folded petals could be stored aboard the Shuttle as a unit and deployed in space as shown in figures 3, 4 and 5. Additional development is required before selection of a reliable mirror coating can be made. The rigid mirror technology pursued in the late 60's resulted in mirrors having specific weights of about one pound square foot. A target specific weight of 0.3 pound per square foot has been established for this application. This goal has been confidently established on the basis of results from an experimental 6-foot diameter mirror designed and fabricated at the Lewis Research Center.

Solar Receiver - the heat receiver absorbs solar radiation from a mirror-collector through an aperture and functions as a heat exchanger to transfer heat into the Brayton system working fluid. The basic heat exchanger portion of the receiver consists of a flux cone made up of 48 gas tubes symmetrically arranged around a center axis to form the frustrum of a cone. Surrounding each gas tube is a second tube containing lithium fluoride which acts as a thermal storage material. During the sun time, excess energy is absorbed to melt the heat storage material. During shade times, the heat storage material freezes as thermal energy continues to be transferred to the working fluid. Thus, the receiver transfers heat continuously to the working fluid which comes from the Brayton system recuperator and delivers heated gas to the turbine for specified maximum shade times. This receiver is designed to operate for as much as 38 minutes in the shade and provide full power to the Brayton system. Doors are provided in order to dump excess heat, especially during operation in orbits with full sun exposure and during reduced power operation. A cross section of this receiver is shown in figure 2. The concept of utilizing LiF as the heat storage material was demonstrated in a full-scale 3-tube test module. This investigation is reported in NASA TN D-6665.

In the design and development of a receiver tailored specifically for the SCB, it is anticipated that a smaller, lighter, more efficient unit could be produced. When a fixed orbit is specified, for example, the heat dump doors would not be required and the correct amount of LiF needed could be specified for that shade time. Investigation would also be conducted on heat pipe designs for the receiver. Using heat pipes instead of the large gas tubes would lead to a smaller, lighter receiver. Figure 6 shows the receiver coupled with the converter.

Brayton Rotating Unit - this unit, referred to as the BRU, consists of a radial inflow single-stage turbine, a Lundell-type alternator and a radial outflow single-stage compressor on a single shaft. The shaft is supported on two journal and one double-acting thrust bearings. These bearings are self-acting gas bearings that utilize the working fluid.

The BRU was designed to produce from 2 to 15 kWe. One of the BRUs delivered has been modified to provide additional cooling to the alternator. With the additional cooling this unit is capable of producing up to 30 kWe. No development problems are known that would impede design and fabrication of a rotating unit for the SCB module.

Recuperator - the recuperator is a single-pass counterflow gas-to-gas heat exchanger with stacked plate-fin surfaces. It has been designed for low gas flow pressure drops and high heat exchanger effectiveness. This unit is also extremely flexible in that it can be optimized for various power levels without a major redesign. This can be accomplished by increasing or decreasing the number of plate fins in the stack.

Heat Sink Heat Exchanger - this heat exchanger employs a cross-counterflow arrangement with plate-fin surfaces in both the gas and liquid flow paths. The liquid coolant, an organic silicone fluid, makes eight passes across the straight-through gas flow.

The recuperator and the heat sink heat exchanger were designed into a single package (BHXU) for the system shown. They can also be designed as separate units with little or no weight penalty.

Electrical Subsystem - the power module electrical subsystem includes a speed control, voltage regulator, a parasitic load, dc power supply, inverters, and a signal conditioner. The BRU has been designed for constant-speed, steady-state operation at 36,000 rpm. To maintain this constant output frequency of 1200 Hertz, a speed control and a parasitic, resistive load are used. The parasitic load absorbs any unused power. The voltage regulator maintains the three-phase voltage at 120 volts, line-to-neutral, or 208 volts, line-to-line. During steady-state operation, the dc power supply rectifies part of the three-phase alternator output to provide the internal needs of the power system at  $\pm 28$  volts. Dc power is used by the engine controls, signal conditioner, and the inverters. The inverters supply 400 Hertz power to the liquid pump motors used in the heat rejection/radiator system.

Item 1

SOLAR BRAYTON SPACE POWER SYSTEM WEIGHT ESTIMATES

| <u>Major Components</u>          | <u>11 kWe<br/>Mid '60's Tech.<br/>Est. Wt., lbs.</u> | <u>11 kWe<br/>Projected Tech.<br/>Est. Wt., lbs.</u> | <u>50 kWe<br/>Projected Tec<br/>Est. Wt., lbs</u> |
|----------------------------------|------------------------------------------------------|------------------------------------------------------|---------------------------------------------------|
| Solar Concentrator               | 370                                                  | 200                                                  | 1500                                              |
| Radiator                         | 1000                                                 | 700                                                  | 4000                                              |
| Connecting Structure             | 210                                                  | 150                                                  | 400                                               |
| Reflector Support Structure      | 70                                                   | 70                                                   | 100                                               |
| Brayton Rotating Unit (BRU)      | 140                                                  | 100                                                  | 150                                               |
| Recuperator/Waste Heat Exchanger | 400                                                  | 350                                                  | 1500                                              |
| Receiver/Heat Storage Unit       | 2000                                                 | 1000                                                 | 4000                                              |
| Parasitic Load and Radiator      | 70                                                   | 70                                                   | 150                                               |
| Electrical Subsystem/Controls    | 300                                                  | 50                                                   | 70                                                |
|                                  | <hr/>                                                | <hr/>                                                | <hr/>                                             |
| Total                            | 4560                                                 | 2690                                                 | 11,870                                            |



Item 5 and 6  
SOLAR BRAYTON SPACE POWER SYSTEM DEVELOPMENT PROGRAM

| Major Task                   | FY | 77 | 78     | 79     | 80     | 81     | 82     | 83     | 84 |
|------------------------------|----|----|--------|--------|--------|--------|--------|--------|----|
| SYSTEM/MISSION DEFINITION    |    |    |        |        |        |        |        |        |    |
| FLIGHT SYSTEM-PRELIM DES     |    |    | 500    |        |        |        |        |        |    |
| COLLECTOR DEVELOPMENT        |    |    |        |        |        |        |        |        |    |
| STOW/DEPLOY MIRROR DES.      |    |    | 100    | 200    |        |        |        |        |    |
| LIGHT-WT MIRROR FAB.         |    |    |        | 1000   | 1000   |        |        |        |    |
| LONG-LIFE MIRROR COATINGS    |    |    | 500    | 1000   | 1000   |        |        |        |    |
| GROUND DEMO. SYS. DES & FAB. |    |    |        |        |        |        |        |        |    |
| ROTATING UNIT                |    |    | 2000   | 2000   |        |        |        |        |    |
| RECUPERATOR/WASTE HX         |    |    | 2000   | 2000   |        |        |        |        |    |
| RADIATOR                     |    |    | 2000   | 3000   |        |        |        |        |    |
| ELECTRICAL SYSTEM            |    |    | 1000   | 2000   |        |        |        |        |    |
| RECEIVER                     |    |    | 2000   | 3000   |        |        |        |        |    |
| SIMULATED SOLAR FLUX         |    |    | 3000   | 3000   |        |        |        |        |    |
| GROUND DEMO. SYSTEM EVAL.    |    |    |        |        | 10,000 |        |        |        |    |
| FLIGHT SYSTEM DESIGN         |    |    |        |        | 5,000  |        |        |        |    |
| " " FAB.                     |    |    |        |        |        | 10,000 |        |        |    |
| " " COMP. QUAL.              |    |    |        |        |        | 5000   |        |        |    |
| FLIGHT SYS ASSY & QUAL TEST  |    |    |        |        |        |        | 15,000 | 10,000 |    |
| FLIGHT UNITS: FAB/INST.-FLT. |    |    |        |        |        |        |        |        |    |
| PROG. MGMT. REPORTS, QUAL.   |    |    | 1,000  | 2,000  | 3,000  | 3,000  | 3,000  | 3,000  |    |
| Totals:                      |    |    | 14,100 | 19,200 | 20,000 | 18,000 | 18,000 | 13,000 |    |

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Recurring Cost: ~ \$2M/Module

Development Costs in \$K

## Item 7

### CURRENT DEVELOPMENT STATUS

The Lewis Research Center has designed, fabricated and tested a closely coupled flight typical 2 to 15 kWe Brayton power conversion system. Performance tests have demonstrated that the system meets or exceeds all performance design objectives. One system has accumulated in excess of 26,000 hours of operation with no measurable degradation of performance. Post-test inspections of the BRU (Ref. NASA TM X-73569) indicate that there is no apparent wear or failure mode that will prevent the attainment of a 5- to 10-year life. Performance testing has demonstrated the accuracy and reliability of sophisticated computer programs used to design and optimize the system and its components.

The only problems encountered in operating this system were leaks that developed in the recuperator. Subsequent design improvements and a fabrication development program have resulted in the fabrication of a recuperator that has been subjected to 200 thermal cycles without a failure.

The technology required to produce a reliable Brayton cycle power conversion loop for space application has been demonstrated by these tests.

A full-scale heat receiver has been fabricated but not tested. Three full-scale tubes were tested and have verified the concept and the heat transfer characteristics. A full-scale radiator was fabricated and tested with the 2 to 10 kWe Brayton engine in a vacuum space chamber with a cold wall to simulate space. The heat transfer characteristics and viability of the concept were verified.

Lewis Research Center has expended about 30 million dollars on the Brayton technology program to date.

## Item 8

### DEVELOPMENTAL RISKS

The known developmental risks identified at this time are in the nature of achieving a useful actual life that meets the required life in the area of mirror and radiator coatings. However, development of alternate coatings and application processes during the course of the program can eliminate this uncertainty.

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### Item 9

#### ESTIMATE THE IMPACT OF ADJUSTING THE POWER MODULE OUTPUT CAPABILITIES LATE IN THE HARDWARE DEVELOPMENT PROGRAM TO REFLECT CHANGES IN SCB POWER NEEDS

The Brayton system offers a great deal of flexibility in adjusting to a wide spectrum of power requirements. Small modules (10 to 15 kWe) in multiple arrays offer the greatest flexibility and also provide redundancy for manned reliability. A great deal of flexibility in adjusting output power also exists within the individual modules. For example, power can be reduced by simply reducing system operating pressure and thermal input. To provide increased power capability, the components can be designed with excess margin with very little weight penalty. Additional power requirements can then be met by increasing system pressure and thermal input. Figure 7 taken from Ref. 2 demonstrates the output power variations available in a fixed set of converter components.

### Item 10

#### ESTIMATE OF MANNED MAINTENANCE OR REPLACEMENT AFTER 5- TO 10-YEAR USE

In view of the exposure of the highly reflective mirror coatings to the hostile (and not fully defined) environment of space, it is anticipated that these coatings will have to be replaced after five years. As noted under "Development Risks", the coating life and maintenance of its high reflectivity is a major developmental task. However, one promising scheme of maintaining the high performance of the mirror coating is to periodically apply a thin layer of aluminum or silver, using an in-situ deposition process. }

Based on the current development status of the Brayton components, it is considered feasible to design and fabricate these components for a 10-year life goal with no maintenance required.

## Item 11

### SYSTEM LIFE AND RELIABILITY ESTIMATES

With the exception of the mirror coatings, the Brayton components may be designed for up to ten years' life with high confidence. Long-life and high reliability were and continue to be keystone objectives of the NASA Brayton technology effort. Careful design of the critical components leads to reduced stresses and high reliability. The desired system life and mission reliability can almost be reduced to a weight trade-off. In unmanned missions, high radiator reliability is achieved by redundant coolant circuits and additional armor. For the SCB application, it appears that the required reliability can be achieved by trade-offs in component redundancy and structural weight versus expending a permissible amount of manned maintenance to the level of simple replacement and leak repairs.

With the adoption of the modular approach to provide additional power, no degradation of life or reliability is anticipated. As discussed elsewhere, survival reliability is improved when a number of similar, independent modules are used. As a point of reference, in the reliability studies on the isotope-powered Brayton system, a reliability number of 0.952 was assigned based on a 5-year life.

## Item 12

### DIRECT THERMAL ENERGY

Direct thermal energy over a wide range of temperatures may be extracted from both the primary gas loop and the secondary liquid cooling loop. For simplicity of control, however, such extraction of thermal energy should be made on a constant, steady-state basis, and consistent with the method of steady-state generation of electric power.

Heat can be extracted for the working gas between the recuperator and the heat-sink heat exchanger (799°R) or from the radiator cooling fluid (741°R - 526°R) with no penalty to system performance because this is nonusable heat. Heat could also be extracted at higher temperatures but this would be usable heat and cycle adjustments would be necessary. Extracting heat from any point in the system would have to be considered and included in the final system design.

### Item 13

#### UNIQUE SAFETY REQUIREMENTS

Prior to recovery, the aperture doors must be closed and the receiver allowed to cool down to assure solidification of the lithium fluoride. After this has been accomplished, the system has no other unique safety requirement either during launch or recovery.

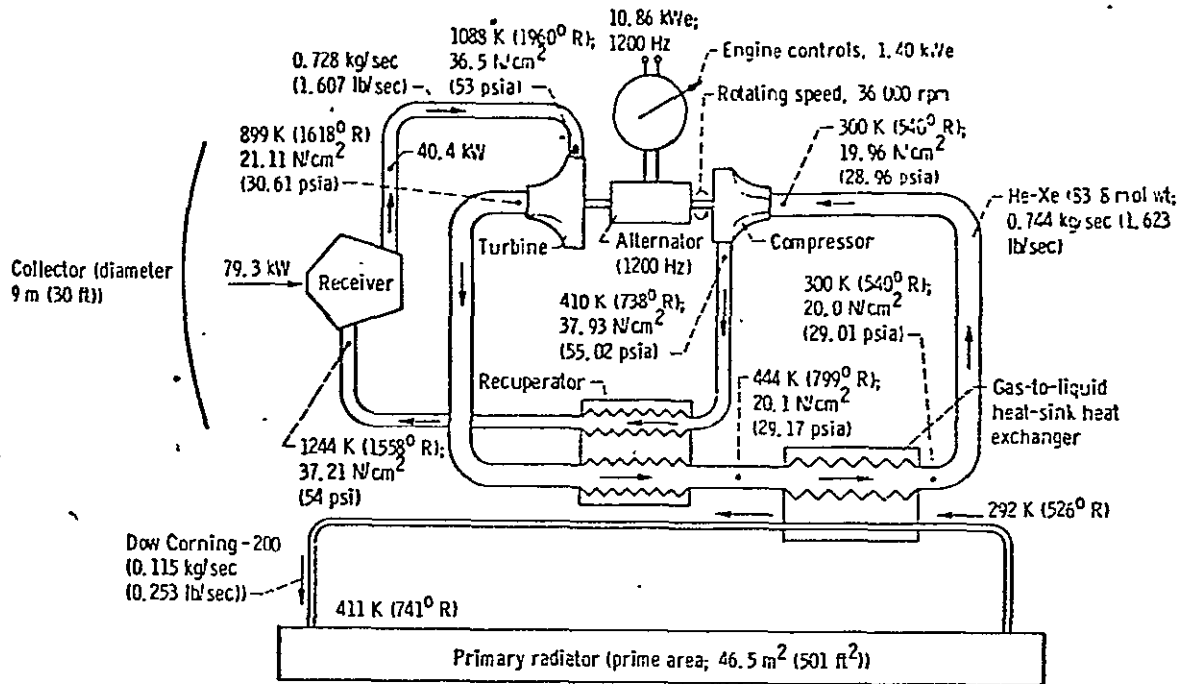
### Item 14

#### FLUID LOOP FREEZING

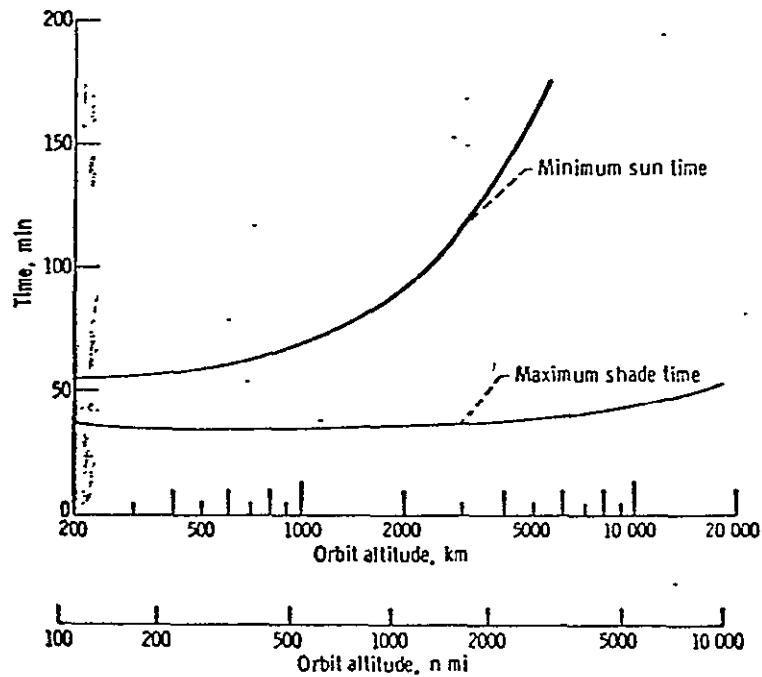
For the coolant loop, a silicone oil with a viscosity of  $2 \times 10^{-6}$  m<sup>2</sup>/sec at 77°F and a pour point at -148°F was selected to preclude freezing in transit or in low earth orbits.

#### REFERENCES

1. Cameron, H. M.; Mueller, L. A.; Namkoong, D.: Preliminary Design of a Solar Heat Receiver for a Brayton Cycle Space Power System. NASA TM X-2552, 1972.
2. Klann, J. L.: Steady-State Analysis of a Brayton Space Power System. NASA TN D-5673, 1970.

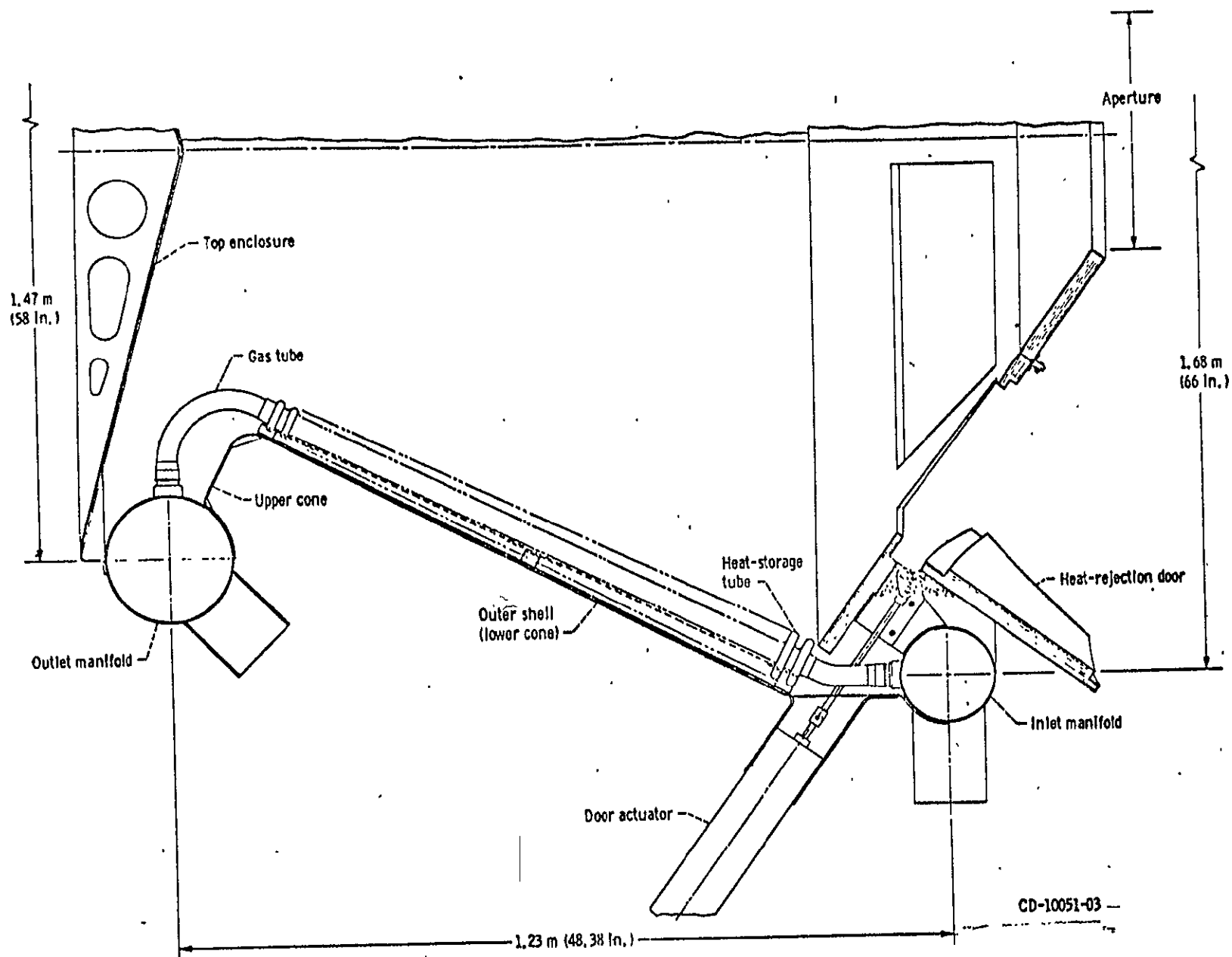


Schematic diagram of solar-powered Brayton powerplant.



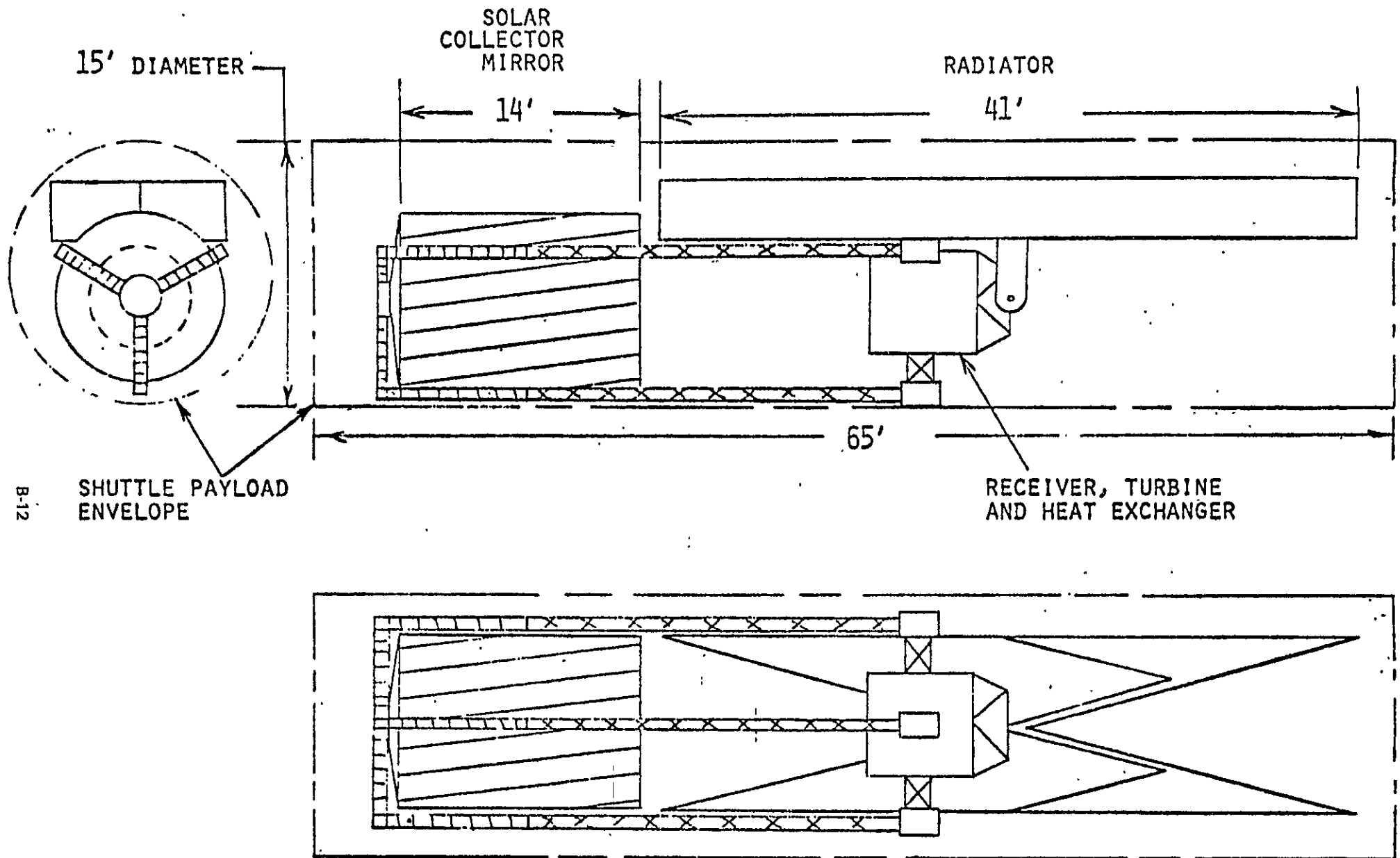
- Sun and shade time as function of orbit altitude.

Figure 1.



Cross section of receiver.

Figure 2.

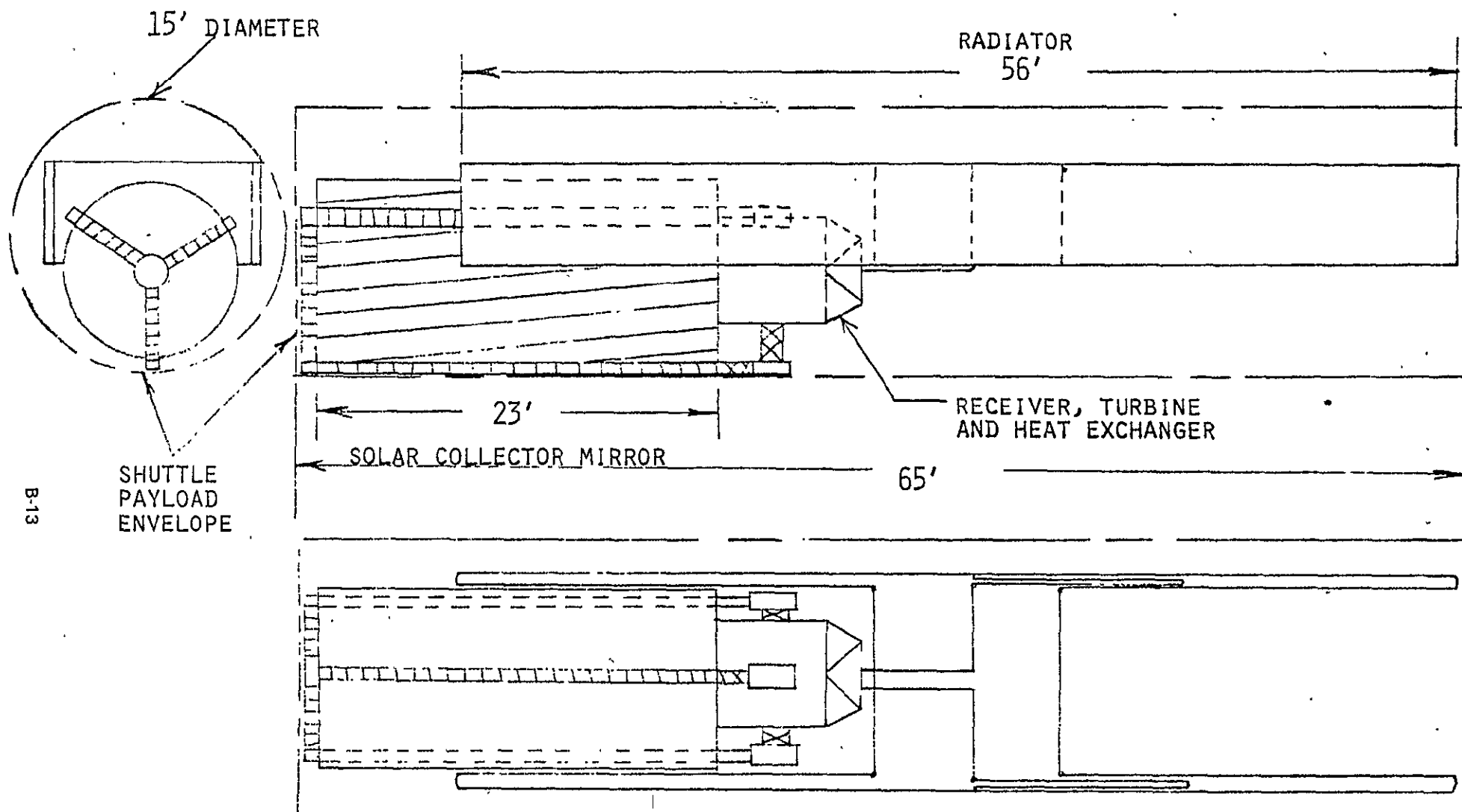


BRAYTON CYCLE EQUIPMENT IN SHUTTLE BAY  
LAUNCH CONFIGURATION

Figure 3.-A (11 kWe)

SCALE  $1/8" = 1'$

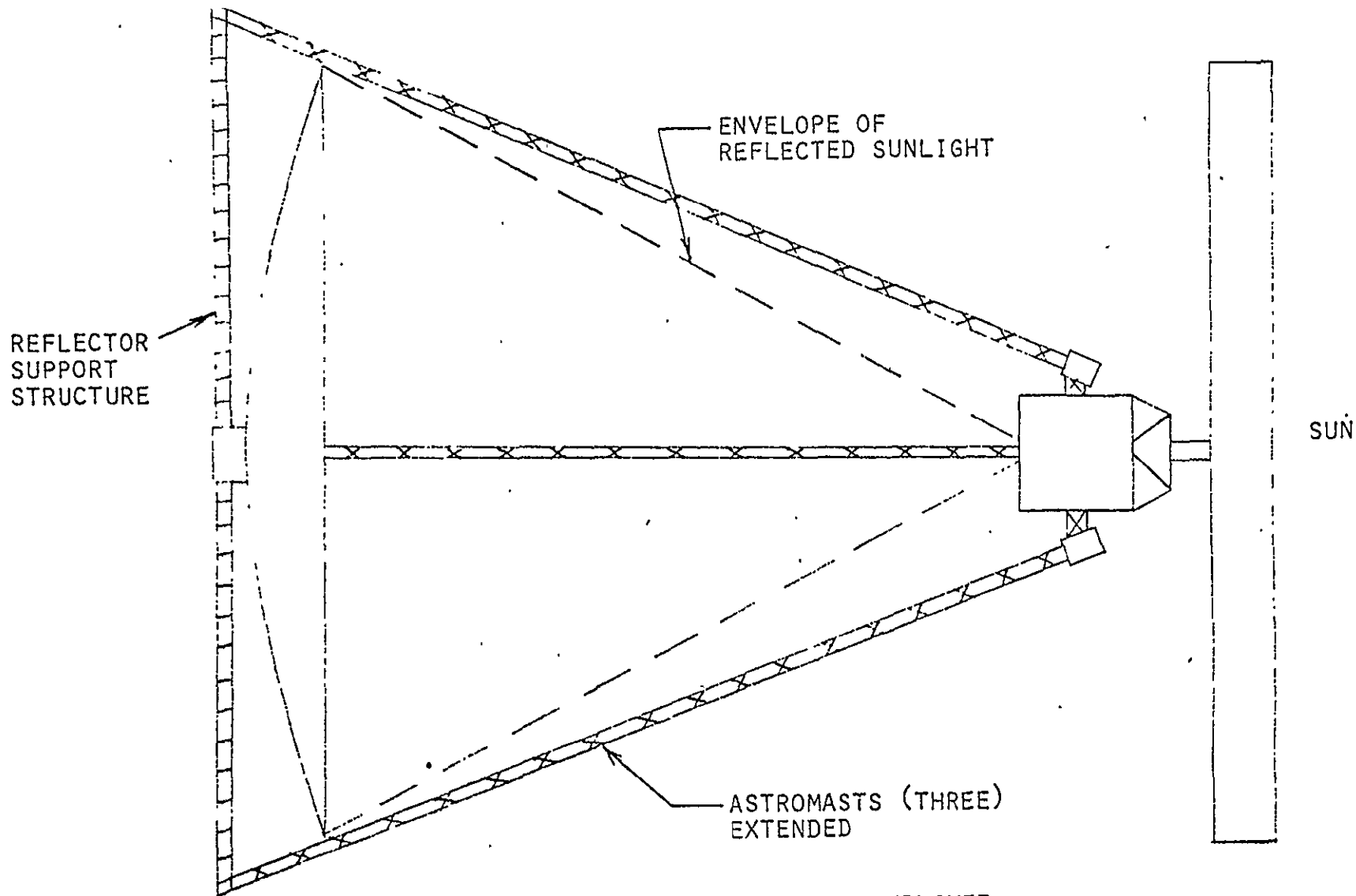




B-13

BRAYTON CYCLE EQUIPMENT IN SHUTTLE BAY  
50 KW MODULE

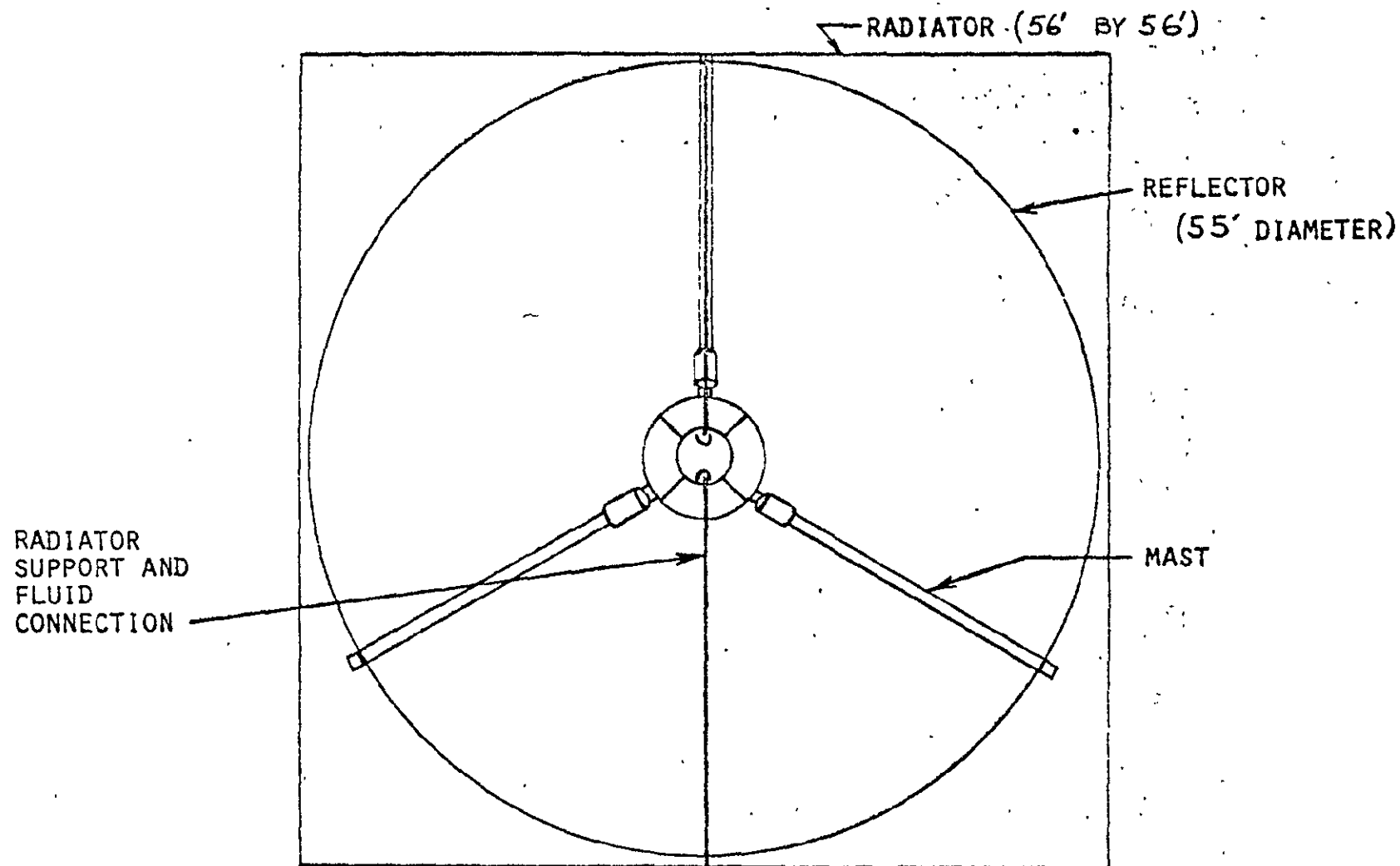
FIGURE 3B



BRAYTON CYCLE EQUIPMENT DEPLOYED

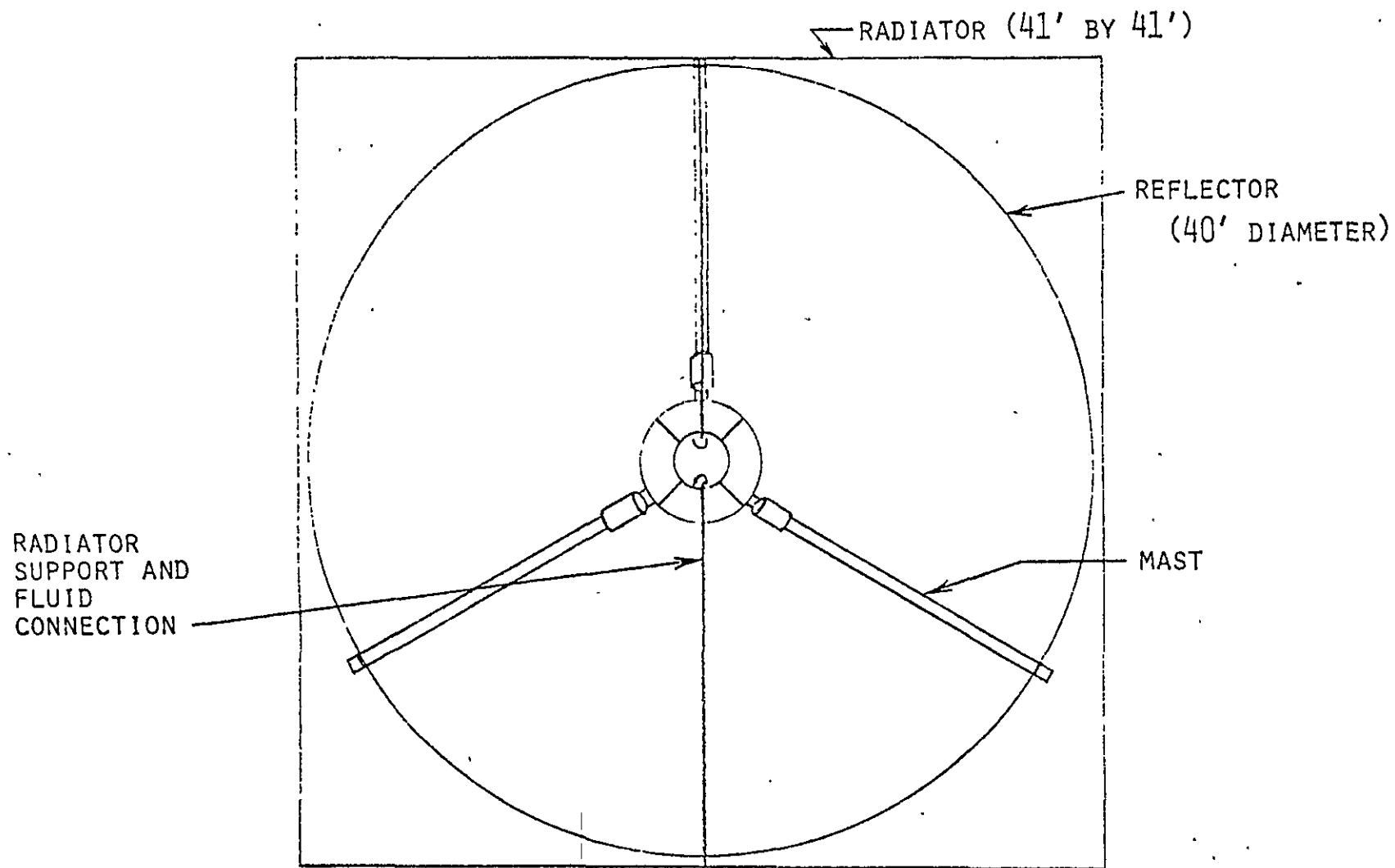
Figure 4.-A (11 kWe)

SCALE 1/8" = 1'



BRAYTON CYCLE EQUIPMENT DEPLOYED  
VIEW FROM RADIATOR END

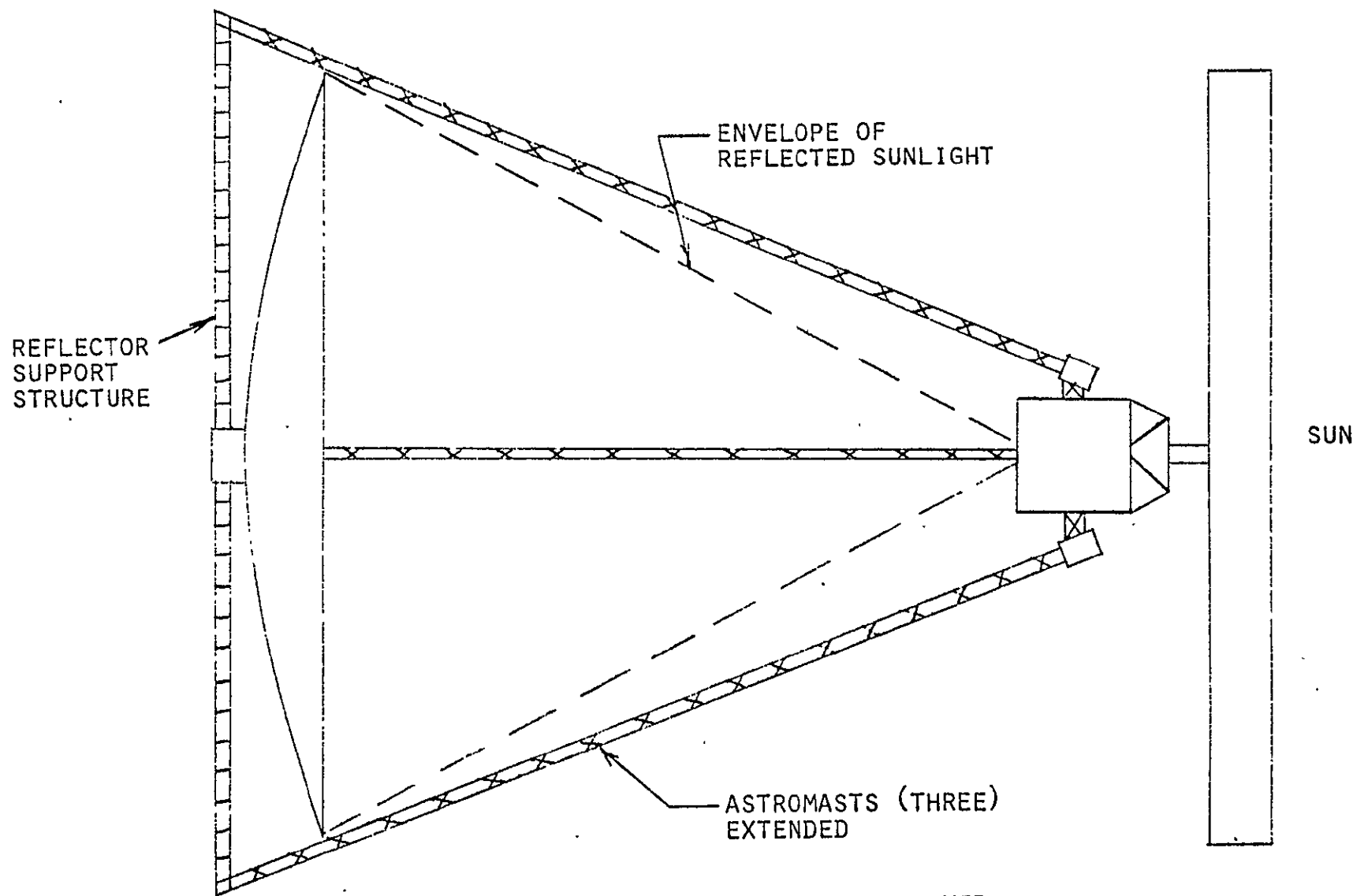
FIG 4B



BRAYTON CYCLE EQUIPMENT DEPLOYED  
VIEW FROM RADIATOR END

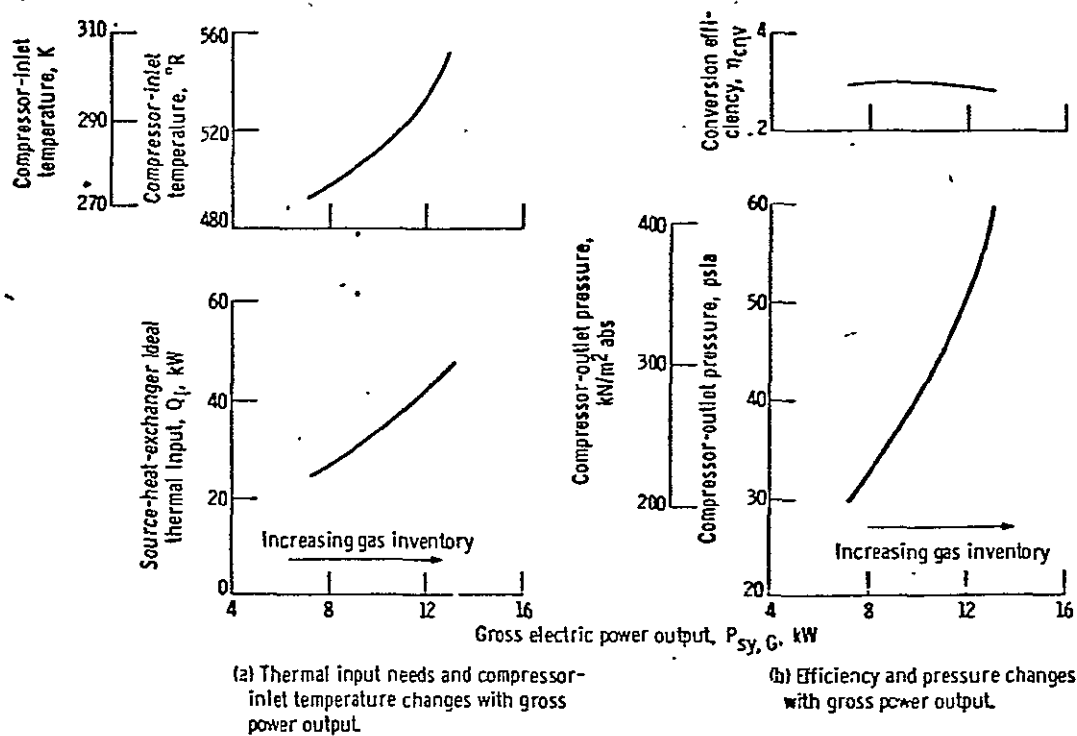
Figure 5.-A (11 kWe)

SCALE  $1/8'' = 1'$



BRAYTON CYCLE EQUIPMENT DEPLOYED

FIG 5B



- Simulated solar operation. Turbine-inlet temperature,  $1960^{\circ} R$  (1089 K); working gas, helium-xenon (molecular weight, 83.8); effective radiator heat-sink temperature,  $400^{\circ} R$  (222 K); prime radiator area, 500 square feet (46.5  $m^2$ ); coolant mass-flow rate, 0.25 pound mass per second (0.11 kg/sec); design average thermal input, 40.5 kilowatts.

Figure 7

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Appendix C  
SOLAR-THERMIONIC SPACE POWER CONCEPTUALIZATION

At an efficiency of 20-22 percent, thermionic power conversion in the 1400K-1600K temperature range appears as a potentially attractive candidate for solar-thermal space power applications. Heat rejection temperature, at 600K-800K, is excellent for efficient space conditioning, for heat engine bottoming cycles, including pumping and control functions, and for compact space radiators. The versatility of the thermionic power conversion in "total energy" concepts is unique when properly applied. A modular power and space radiator system lends itself readily to those concepts for manned habitats and work stations.

I. Solar Concentrator and Receiver

A key technology requirement for high-temperature solar-thermionic systems is a precision parabolic solar concentrator and receiver. Concentrator reflecting surface accuracy, stability and lifetime, together with concentrator pointing accuracy, are quite difficult to achieve. But with recent advances in lightweight, composite structural materials, and in computer design of structures, economically attractive new approaches are now offered to an old problem. In addition to a basic deployable structure, a precision parabolic surface substrate and reflecting mirror are required.

For a depolyable structure, it appears that excellent technology has already been developed although it was done for large furlable antennas for the early 1980's (Ref. 1). Further development, explicitly for solar collectors, may offer low-cost alternatives if time allows, but is not considered here. Other possibilities also exist for erectable systems, such as the Boeing antenna concept of Figure 1, which require significant EVA by the Shuttle crew. These antenna systems require approximately an order of magnitude higher accuracy than will be needed by the solar collector and are quite costly.

The wrapped rib structure concept (Ref. 2) was tentatively selected here because of its extremely high packaging efficiency. This concept consists of a hollow donut-shaped hub attached to a series of radial ribs cut to the shape of a parabola. Flexible tension lines are to be stretched between the ribs and will serve to hold a stiff parabolic surface substrate in place. Figure 2 is an artist's sketch of this concept for the Large Deployable Antenna Shuttle Experiment (LDASE). For solar-reflector transport in the Shuttle, the axis of the hub would be parallel to the axis of the Shuttle cargo bay, perpendicular to that shown in the Figure 2 experimental setup. This entire array can be stowed in a single Shuttle pallet for deployed diameters of up to 100 meters. Active evaluation and control capability is available as an EVA adjustment to periodically measure and readjust the ribs by elevation and/or rotation for a more-accurate surface. Cost of a 30-m unit is presently approximately \$5 million, with an additional two million dollars for active evaluation and control. Costs could be lower by 1984, particularly for solar concentrator applications.

The precise parabolic reflector substrate concept for the solar collector is a unique, cost-effective and innovative concept proposed by JPL (Ref. 3). Stiff mirror segments are to be mounted directly to a graphite-epoxy or metal monocoque structural panel by kinematic detail. This is accomplished by cementing a pattern of fasteners to the back of the mirror segments. These fasteners will hold the mirror segments to the panel surface in a way that will eliminate strain that might be caused by differential expansions. The diagram of Figure 3 illustrates this concept. The substrate makes use of structural design concepts developed for spacecraft photovoltaic solar panels, but with emphasis placed on low-cost fabrication techniques. The panels are to be fabricated for tongue-and-groove interlocking for very simple EVA attachment to the rib structure. Held in place by tension members, disassembly and restowage in Shuttle is also a simple process. Development of the substrate technology will require approximately two years and a cost of about \$1 million, much of it to develop tooling and process controls. Cost of a set of panels, with mirror segments, for first flight is roughly estimated at \$2 million, including test and integration.



A thermionic receiver module analyzed for terrestrial systems at 20 kWe is shown in Figure 4 (Ref. 4). Development costs of the spacecraft receiver (less thermionics) will be approximately \$100,000, with the recurring cost expected to be about \$10/kWe. Several different receiver designs are already progressing for several different power conversion concepts for terrestrial and space power.

The basic figure of merit of a solar concentrator is its concentration ratio. This is defined as the ratio of the concentrator area to the receiver area. Three basic elements contribute to this concentration ratio (Ref. 4). The first element is focal length. Because the sun's image is finite, approximately 0.5 degrees, maximum theoretical concentration ratio is set by the selected focal length of the system (see Figures 5 and 6). At focal length of 0.6, for instance, maximum concentration ratio is approximately 13,000. The second element that establishes concentration ratio is figure error. This is the measure of the concentrator mirror surface deviation from the theoretically-perfect parabolic shape, and is defined as an angular error, or "slope" error. The third element of concentration ratio is the solar tracking error, or the inaccuracy of the control system in tracking the sun. Both the tracking and figure slope errors, measured in degrees, require that the receiver aperture be enlarged in order to accept all the direct rays of the sun reflected from the concentrator. The concentration ratio is thus decreased, as shown in Figure 6, as a function of tracking and figure errors.

The efficiency of energy collection for any given concentration ratio is a function of the concentrator surface. This includes reflectivity of the reflecting surface and the refractive and reflective characteristics of the protective layer over the reflecting surface. The collection efficiency also falls off with increasing receiver temperature. Dependent on cavity absorptivity and emissivity, a finite aperture area will allow direct thermal radiation loss to space. Such loss increases as the fourth power of the temperature. Figure 7 is a plot of theoretical collection efficiency as a function of temperature for the range of concentration ratios of interest.

This figure uses a silver reflector, protected by a 1.27 mm (0.05 in) protective cover glass. It should be noted that, for a given efficiency, a higher concentration ratio will allow operation at a higher temperature.

## II. Thermionic Power Conversion

The two principal factors in the selection of a thermionic power converter are its efficiency and power density. Each of these varies as a function of emitter operating temperature and converter barrier coefficient as shown in Figures 8 and 9. Collector temperature is approximately half the emitter temperature. Some test data are shown on these charts, together with some expected 1977-1981 research goals.

The present "diode" technology requires high-temperature operation in order to achieve a reasonably-high efficiency at high power density. Research indicates that new, low-work-function collector materials may soon be available to achieve higher efficiencies, but with two drawbacks: (1) temperature must be reduced to prevent materials sublimation and back-emission problems and (2) power density may be extremely low. This means that, in both instances, mass/efficiency of the system is adversely affected. Work function and efficiency can be improved by introducing alternate modes of operation that will decrease the plasma losses in the converter. In some cases this requires moving away from typical "diode" configurations. Such alternatives are now being tested in the laboratory, and they indicate that high power density may be achievable at an efficiency of perhaps 20% or more. Mass/efficiency minima appear to favor this alternative. Good results in the temperature range of 1400K-1600K would be desirable, and appear to be only one to two years away.

It should require approximately three to five years after R&D demonstration of technology to develop a flight-ready converter. Thus, if we assume a first test flight in 1984-1986, a technology cutoff date of 1981 or before is needed. By this date our best estimate of thermionic converter performance is a barrier index,  $V_B$ , of 1.4 for an emitter operating temperature of 1400K. This corresponds to a converter operating efficiency of 20% and a power density of  $5.5 \text{ We/cm}^2$ . At 1600K, a barrier index of 1.5 should

be achievable, for a 22% efficient converter with a power density of  $9.2 \text{ We/cm}^2$ .

The above assumes that collector temperature (and heat rejection temperature) is unconstrained. However, this is not necessarily true if the thermionic converter is to be used for power topping applications. When both the emitter and collector temperatures are constrained, thermionic converter efficiency may be expected to vary approximately as shown in Figure 10. In all probability, of course, different materials will be used for different temperature regimes, so that in practice the curves are not necessarily smooth nor totally continuous.

### III. System Parametric Evaluation

Several different elements enter into a system evaluation, including waste heat radiator, electrical cabling, power processing, environmental protection, etc. For a manned space construction base application it is also important to at least partially decouple the precise attitude control and stabilization of the solar-thermionic system from the rest of the manned operations. Detailed studies will be required of this interface as more becomes known about the requirements and constraints of the application. For the moment, such an interface will simply be assumed to exist. The heat rejection system is assumed to terminate in a waste heat radiator.

Radiator size and mass increase rapidly as temperature decreases. Specific size (per unit power) is shown in Figure 11 as a function of temperature and power conversion efficiency. Thermionic power output is low voltage d.c. so that electrical cabling and power processing are relatively heavy. If high-temperature electrical isolation is provided to allow series interconnection of thermionic converters up to at least  $\pm 25 \text{ VDC}$ , power inverters and low-voltage buses and cables should be available at approximately  $5 \text{ kg/kWe}$  and 95% efficiency, based on a multiphase solid state concept available.

The space environment of the solar thermionic power plant will require considerable study. At 270 nautical miles or higher, there will

possibly be significant proton radiation as well as UV radiation. If there is a large concentration of manpower and materials in orbit, a sizeable increase of contamination could occur locally, requiring protective coatings for refractory materials and optical surfaces. Plasma-sprayed and surface-fired ceramics are available, but self-healing oxidation-resistant coatings are not always compatible with a vacuum environment. Meteoroid environment is sensitive to orientation and ephemerides, where considerable variation is possible for approaches to protection.

The adverse characteristics of solar occultation in LEO are apt to make life very difficult for solar thermal systems generally, and for thermionic power conversion particularly. The solar concentrator becomes a very efficient thermal radiator during occultation. Thus, the receiver must be decoupled from the collector during these periods of occultation. Surface accuracy of the collector is also apt to drift significantly with frequent thermal cycling. At the high thermionic operating temperatures, alkali liquid metal working fluids are utilized. NaK, if it is used in the heat rejection loops, will freeze at approximately 300K, and must be maintained above this temperature during occultation. Other alkali metals have higher freezing temperatures, but are not hampered by problems of freezing. Heat pipe radiators, for instance, can be designed to avoid startup problems that might cause system failure. Thermal coefficients of expansion are not expected to be of great concern with the concentrator-receiver design concept proposed here. The special collector mirror segment hold-down method eliminates strain, as does the tongue-and-groove joining. The mirror protective coating also acts as an effective lubricant toward any differential motion. In every respect it appears that the monocoque panel-mirror assembly is a simple EVA assembly task that should also achieve a low-cost production design.

Alkali metals represent a potential explosive safety hazard in case of an abort that results in impact in water. Jettison of the receiver with its thermionic and heat pipe radiator modules will probably be necessary for these special cases. These modules should therefore be designed as a separate, expendable unit.

Combined performance of solar concentrator (Figure 7) and thermionic converter (Figure 10) is shown in Figure 12. For concentration ratio of 2000-3000, optimum receiver temperature appears to be 1400K to 1500K, with heat rejection temperature between 650K and 800K. With power conversion efficiency at 20 percent or slightly above, heat rejection radiator size from Figure 11 shows approximately  $0.2 - 0.45 \text{ m}^2/\text{kWe}$ . A cylindrical heat pipe radiator will have a specific mass of approximately  $0.5 - 1.0 \text{ kg/kWe}$  over the heat rejection temperature range.

#### IV. System Description

Because of desired early scheduling for flight, technology has been selected based on only two years of R&D, followed by design and fabrication of a system proof test model and initiation of flight hardware programs. Selection starts from an unperturbed present research program. This allows a high probability of success, since requirements for engineering breakthrough are virtually eliminated. In all probability there is at least a 50% chance of exceeding the performance estimates of this study.

A nominal 100 kWe power system is proposed at this time, but with recognition that actual power level may be anywhere between 10 kWe and 150 kWe. The largest R&D cost of the solar-thermionic system is associated with the thermionic power conversion, power processing and heat rejection assemblies. These will require approximately \$5 million over a two year period. Such a program would lead to producible 5-10 kWe modules with a total specific mass of approximately  $10 \text{ kg/kWe}$ . This would be followed by a three year development and qualification of PIM and flight hardware at a cost of approximately \$20 million. Subsequent recurring cost of hardware, after the first flight, should be approximately \$1000/kWe, with additional QA&R and spacecraft integration that may be as high as \$4000/kWe.

The large furlable antenna structure is already designed for a 31 m (100 ft.) diameter, suitable for a 150 kWe system. It is also available at smaller sizes. Scaling factors for the solar receiver will be evaluated during

the initial R&D effort and properly integrated with the modular power conversion, radiator and power processor elements. The primary problem with power scaling is the fabrication of tooling for the precision parabolic reflector and substrate, which will require approximately six months and \$500,000 for fabrication and qualification after initial R&D has been accomplished.

Diameter of the 100 kWe parabolic concentrator is approximately 25 m (80 ft.). Its mass is under 2000 kg. Since there is considerable question of materials stability and interlocking accuracy for precision surfaces, the wrapped rib structure will initially be designed for active evaluation and control of the ribs in both the elevate and rotate directions. However, because of the high cost of this control feature, careful studies should be carried out to determine whether such design is necessary. The furled wrapped rib structure is mounted in the shuttle with its axis parallel to the longitudinal axis of the cargo bay. The receiver, with its thermionic converters, radiators, inverters and cabling, is mounted at the focal point position and supported for launch. Mass of this assembly is 1000 kg or less. The maximum required radiator area is 20-45 m<sup>2</sup>, which, for a cylindrical radiator 4.5 m in diameter, requires approximately 3 m of cargo bay length. The inverters, operating at 400K temperature, require a radiator area of 4 m<sup>2</sup>, or an additional 0.3 m of Shuttle bay. Receiver requires approximately 1.25 m, nested inside the cylindrical radiator assembly (Figure 13). These assemblies attached to the furled wrapped rib structure require approximately 7 m of the Shuttle bay, leaving approximately 10 m for the tongue-and-groove jointed parabolic substrate stacks overlaid with their polished mirror segments (with cover glasses if needed to avoid contamination or damage of reflective surfaces).

The array is to be inertially oriented in space except for gravity-gradient torques. Typical gimbal axes and actuators are provided, but there is some question whether this is charged to the solar array or to the spacecraft. Requirements are that the array be oriented to the sun within 0.1 degree or less. However, mass is not exceptionally large and angle changes are relatively slow. Two-axis gimbaling can be handled by redundant pairs of digitally operated stepper motors mounted through a gear train on each

gimbal axis (Ref. 5). Appropriate mechanical disconnect is needed in case of component failure. Direction of motion and torquing directions with respect to gravity gradient torques, orbit inclination and solar alignment must all be studied in detail in future studies. Attitude control instrumentation, gimbals and actuators have an estimated cost of \$3 million for development for flight, with a recurring cost of roughly \$300,000.

Mass and size estimates include approximately 20% redundancy for operation unattended for approximately 10 years. As noted above, however, adjustable control of structure has been assumed to assure maintaining parabolic accuracy consistent with a concentration ratio of 2000 to 3000. Such adjustment may be required as often as 10-15 times annually. With careful mechanical design, there is significant expectation that even this adjustment is unnecessary. With good fabrication tooling for the parabolic substrate, reflective surface accuracy will allow RMS mismatch at the interlocking sections of 0.5 - 1 mm (20-40 mils). This is an order of magnitude less severe than the current antenna design.

Total system non-recurring cost is estimated at \$48 million, plus \$0.5 million and six months if power level cannot be defined until late in the development program. Recurring system cost, after first flight, is \$7.8 million. No maintenance is anticipated over the ten year life.

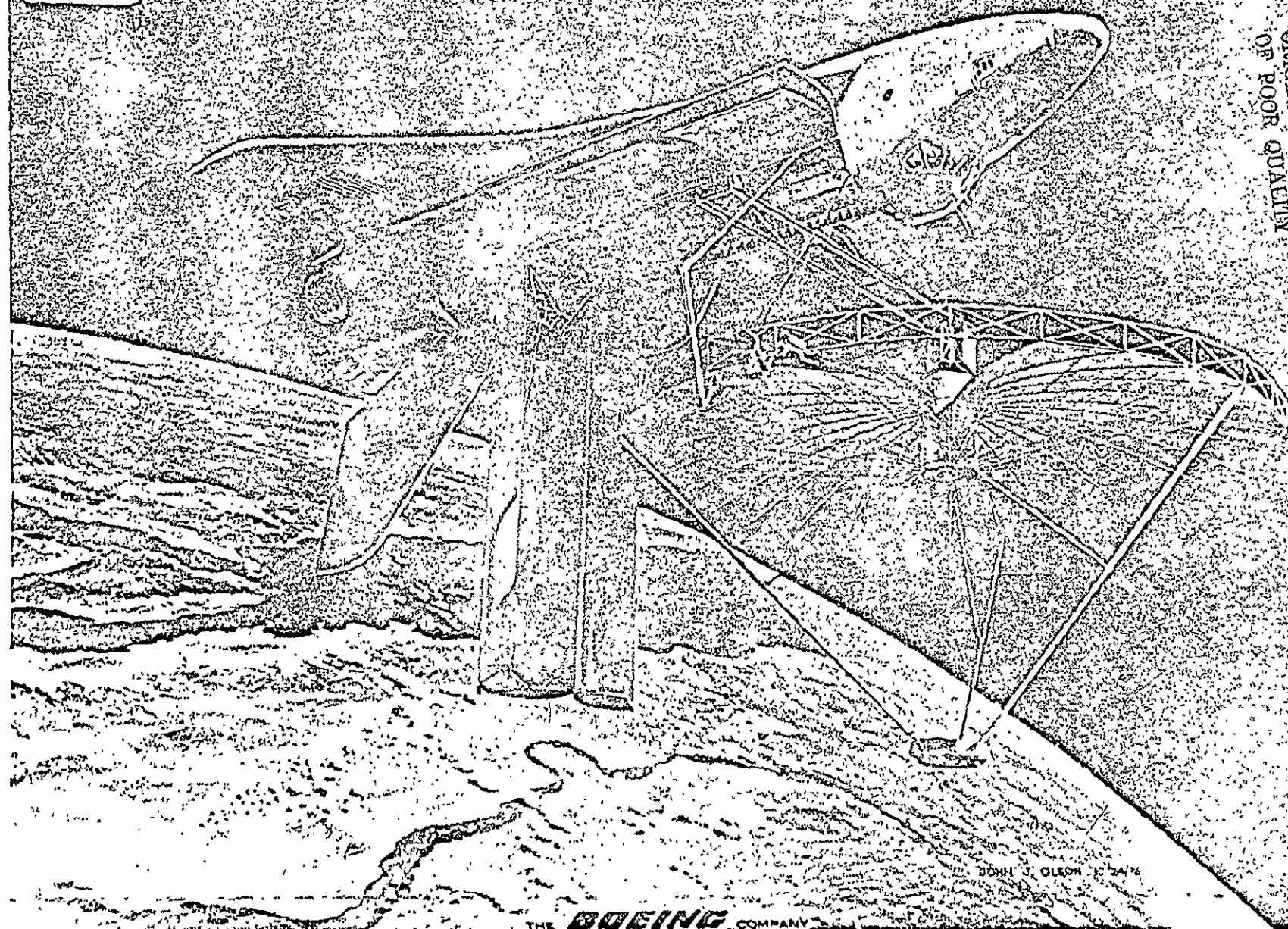
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2. Large Furlable Antenna Study, JPL Contract No. 954082, Lockheed Missiles and Space Company, Jan. 20, 1975.
3. Perkins, G.S., "New, Low-Cost Parabolic Solar Mirror Panel", JPL FY'77 Director's Discretionary Fund Proposal, October, 1976.
4. Swerdling, M., et al, "Feasibility Study of a Solar Thermionic Power System for Terrestrial Applications", JPL Internal Report No. 900-719, December, 1975.
5. Perkins, G.S., "Viking Orbiter 1975 Articulation Control Actuators", Proceedings, 8th Aerospace Mechanisms Symposium, NASA Langley Research Center, October 18-19, 1973.



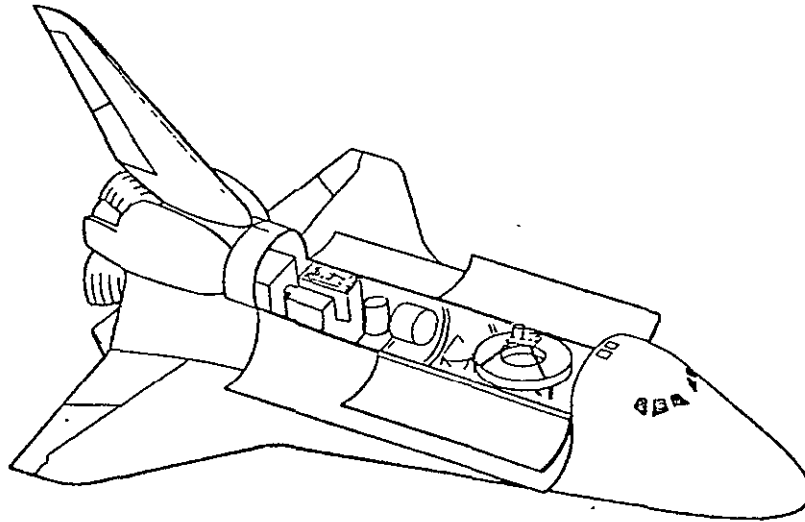
FIGURE 1

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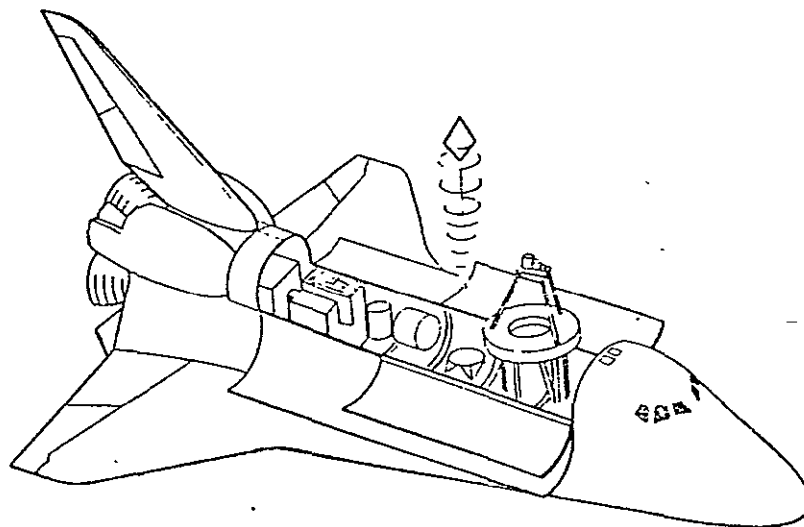


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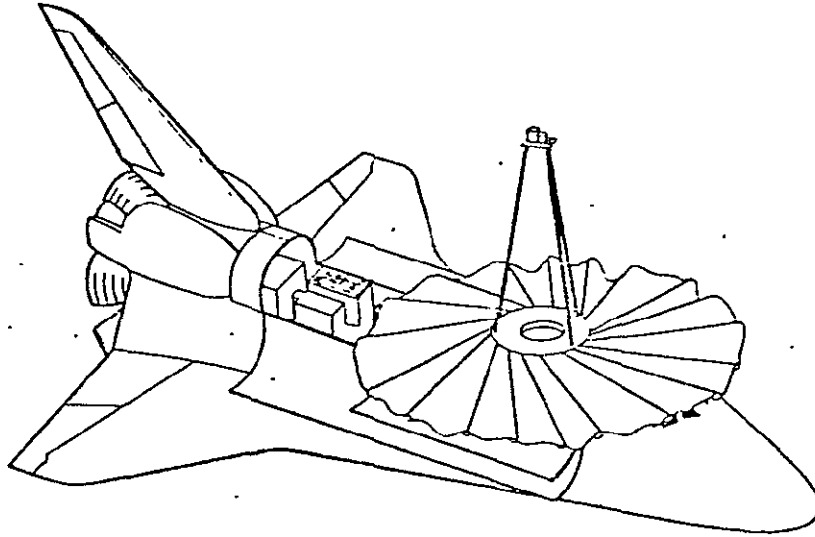


(a) LDASE Launch Configuration

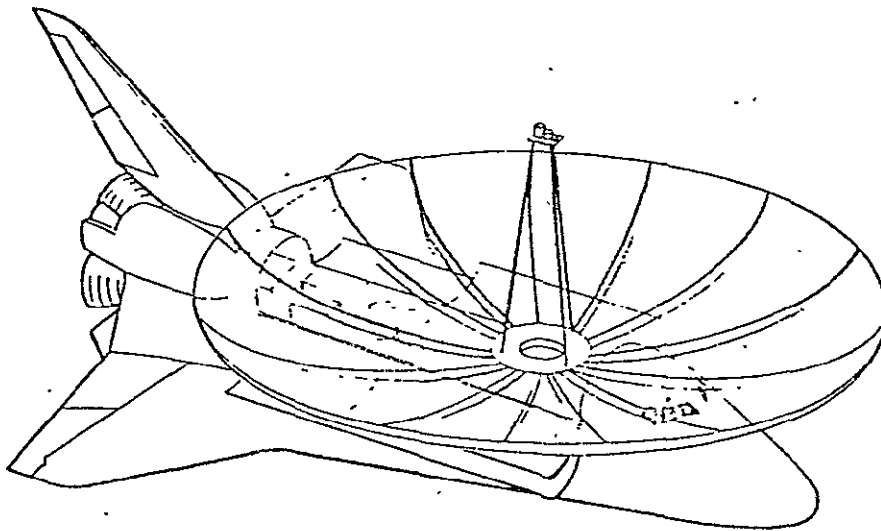


(b) Beacon Separation, Feed Structure Deployment  
and Elevation of Stowed Reflector

Figure 2. Artist's Conception of the Mechanical Configuration  
and Deployment Sequence for the LDASE



(c) Partial Reflector Deployment



(d) Fully Deployed LDASE Configuration

Figure 2. Artist's Conception of the Mechanical Configuration and Deployment Sequence for the LDASE (contd)

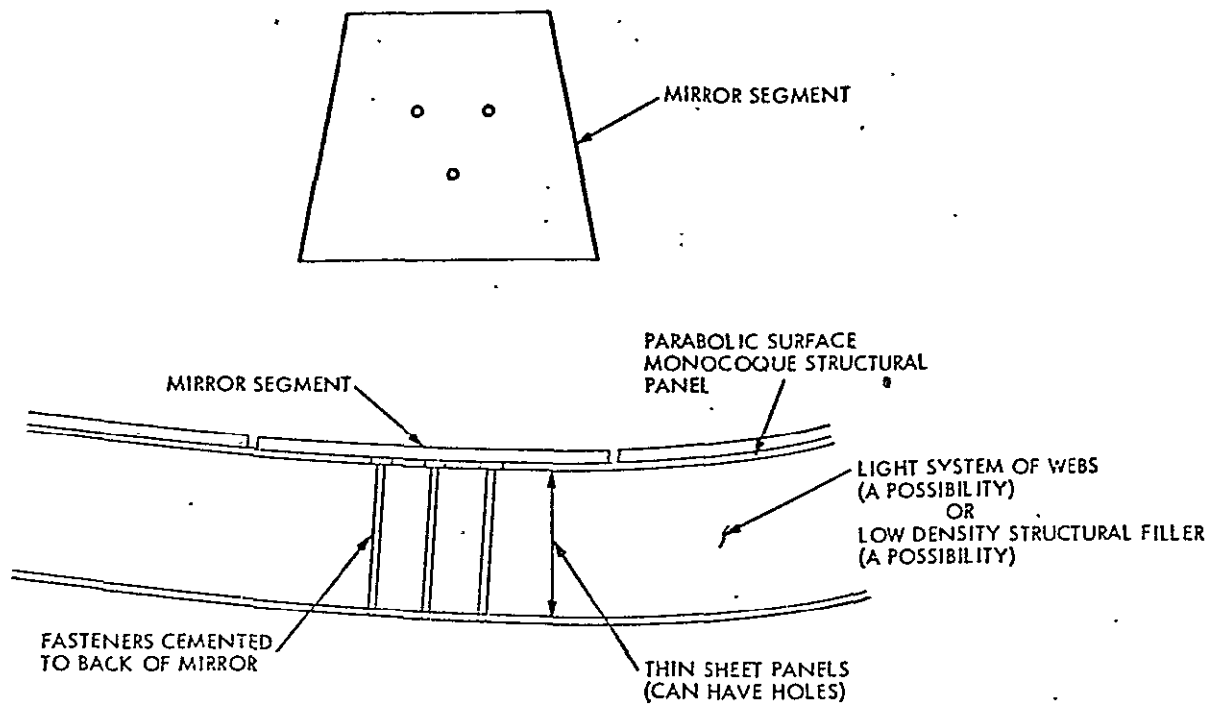


Figure 3. Parabolic Reflector Substrate Concept with Mirror Segments

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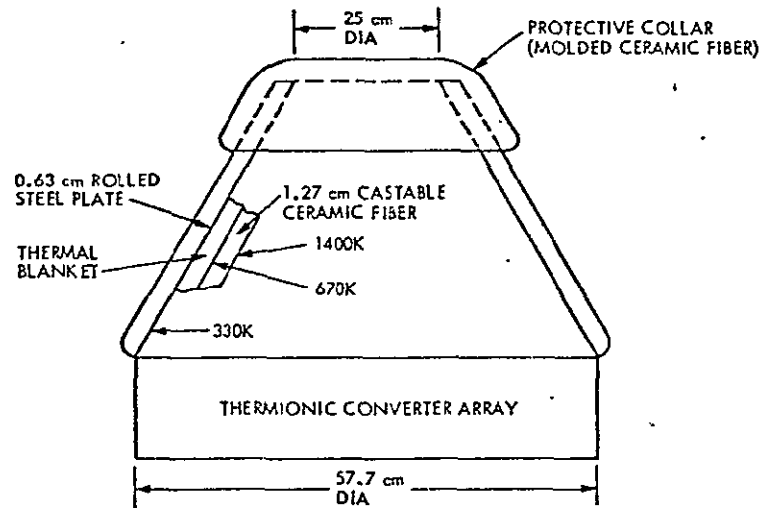


Figure 4. Thermionic Receiver/Module Baseline Configuration, 20 kWe (Terrestrial)

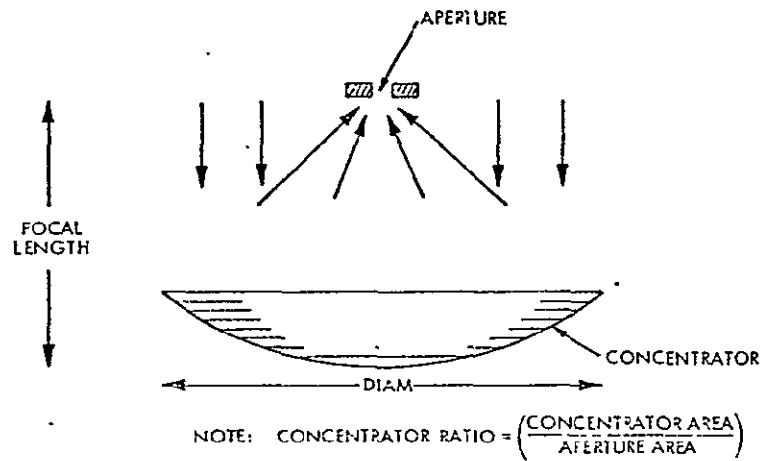


Figure 5. Solar Concentration Parameter Definition

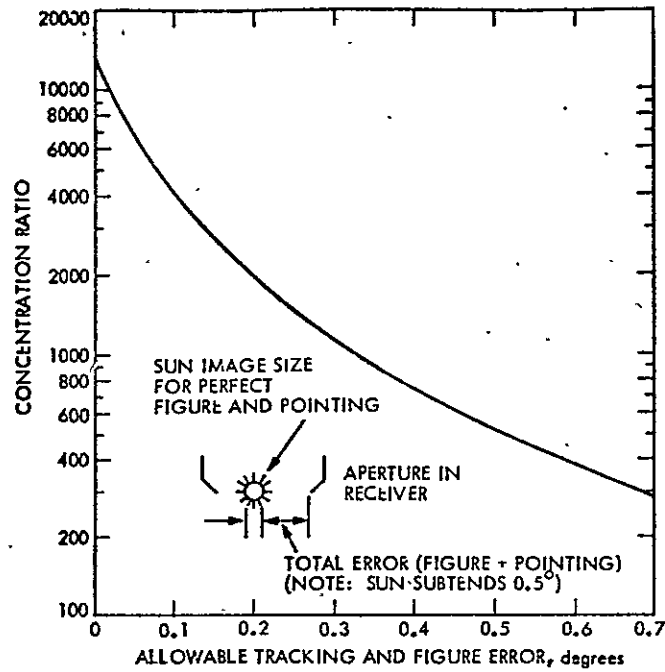


Figure 6. Concentration Ratio vs Allowable Tracking and Figure Error

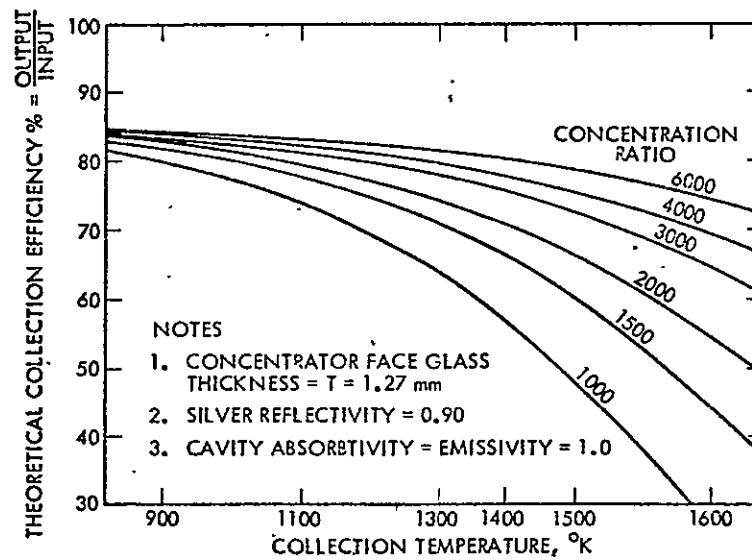


Figure 7. Theoretical Collection Efficiency vs Collection Temperature

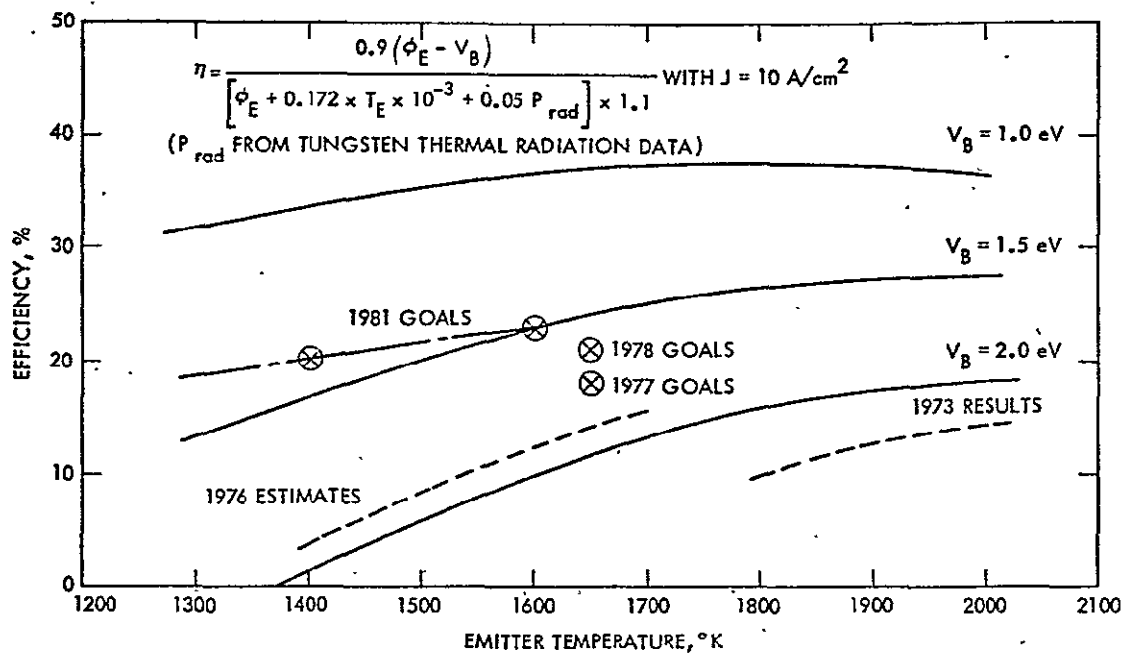


Figure 8. Efficiency vs Emitter Temperature

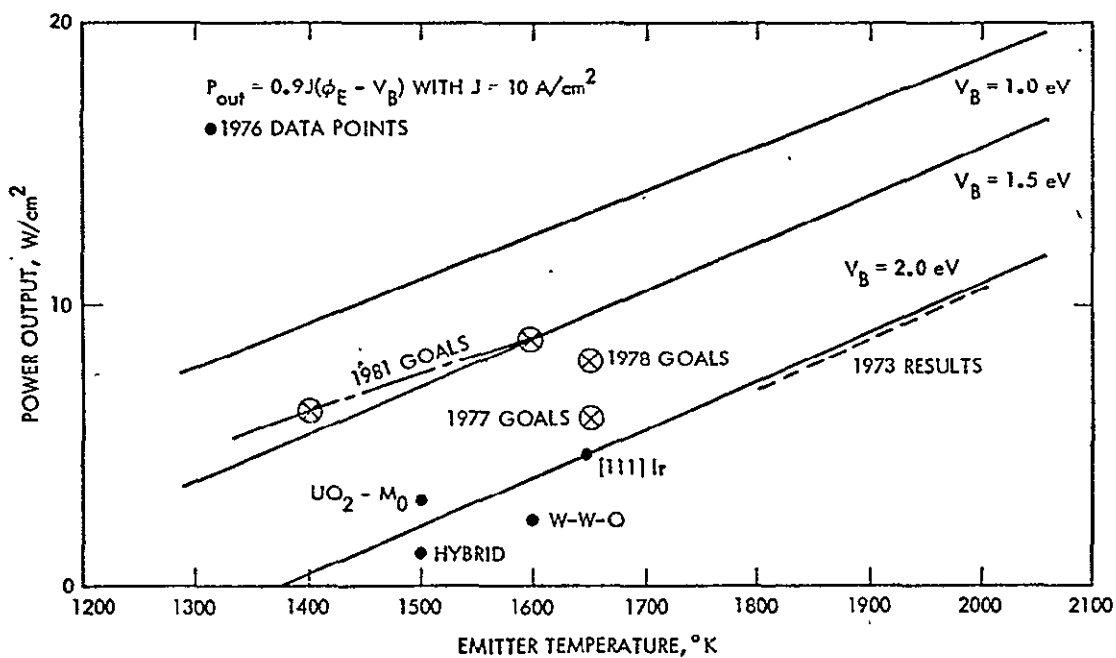


Figure 9. Power Output vs Emitter Temperature

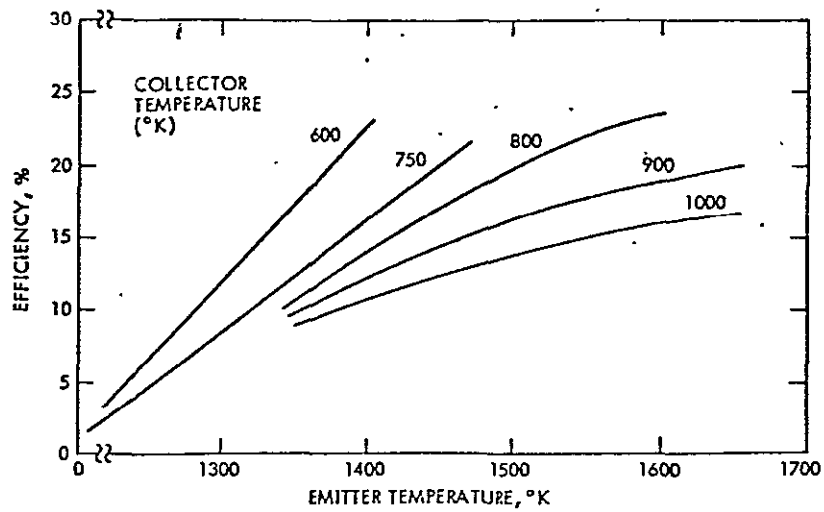


Figure 10. Estimated Converter Efficiency with Collector Temperature Constrained

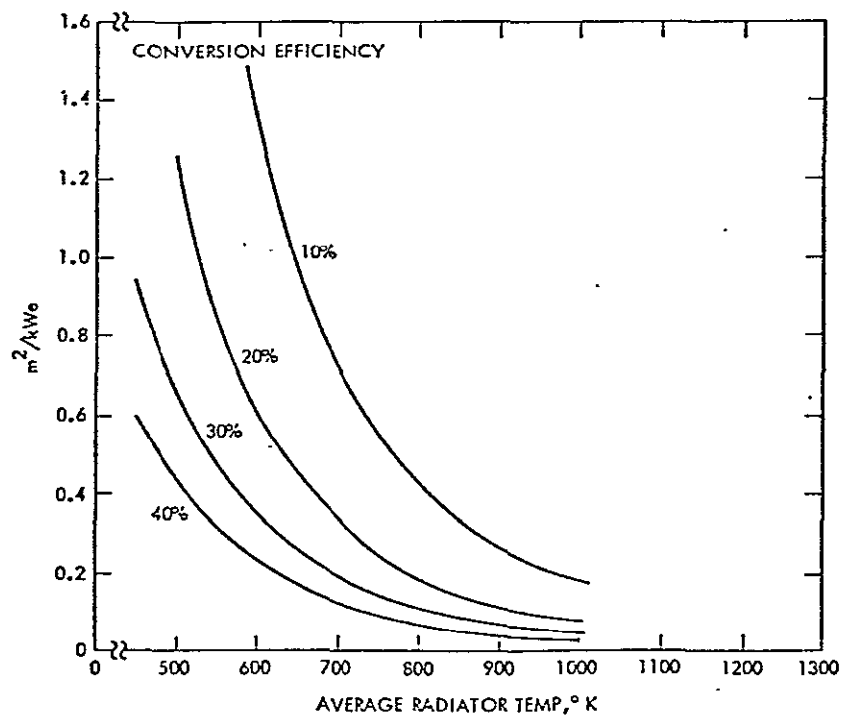


Figure 11. Radiator Specific Size



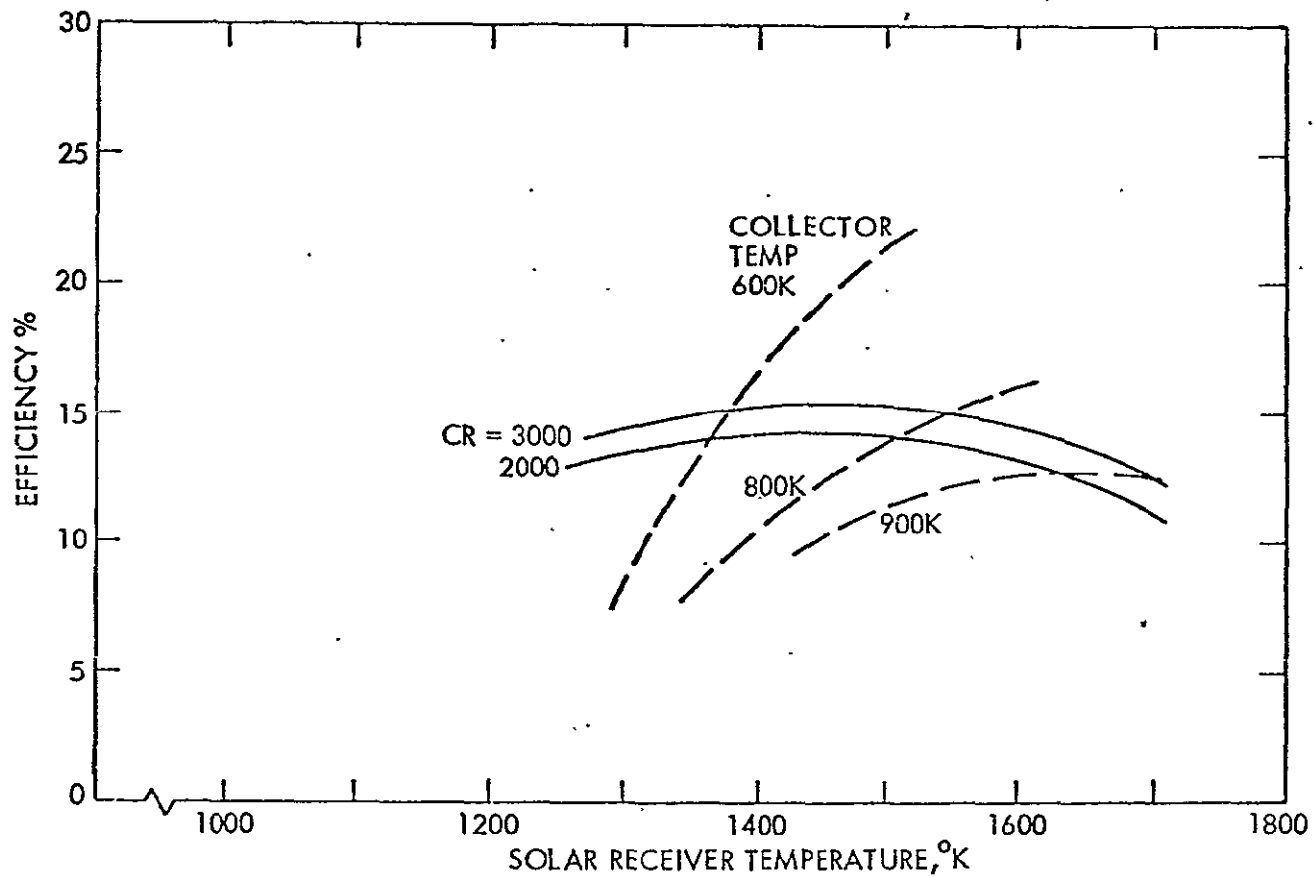


Figure 12. Combined Efficiency of Solar Concentrator and 1981 Thermionic Converter

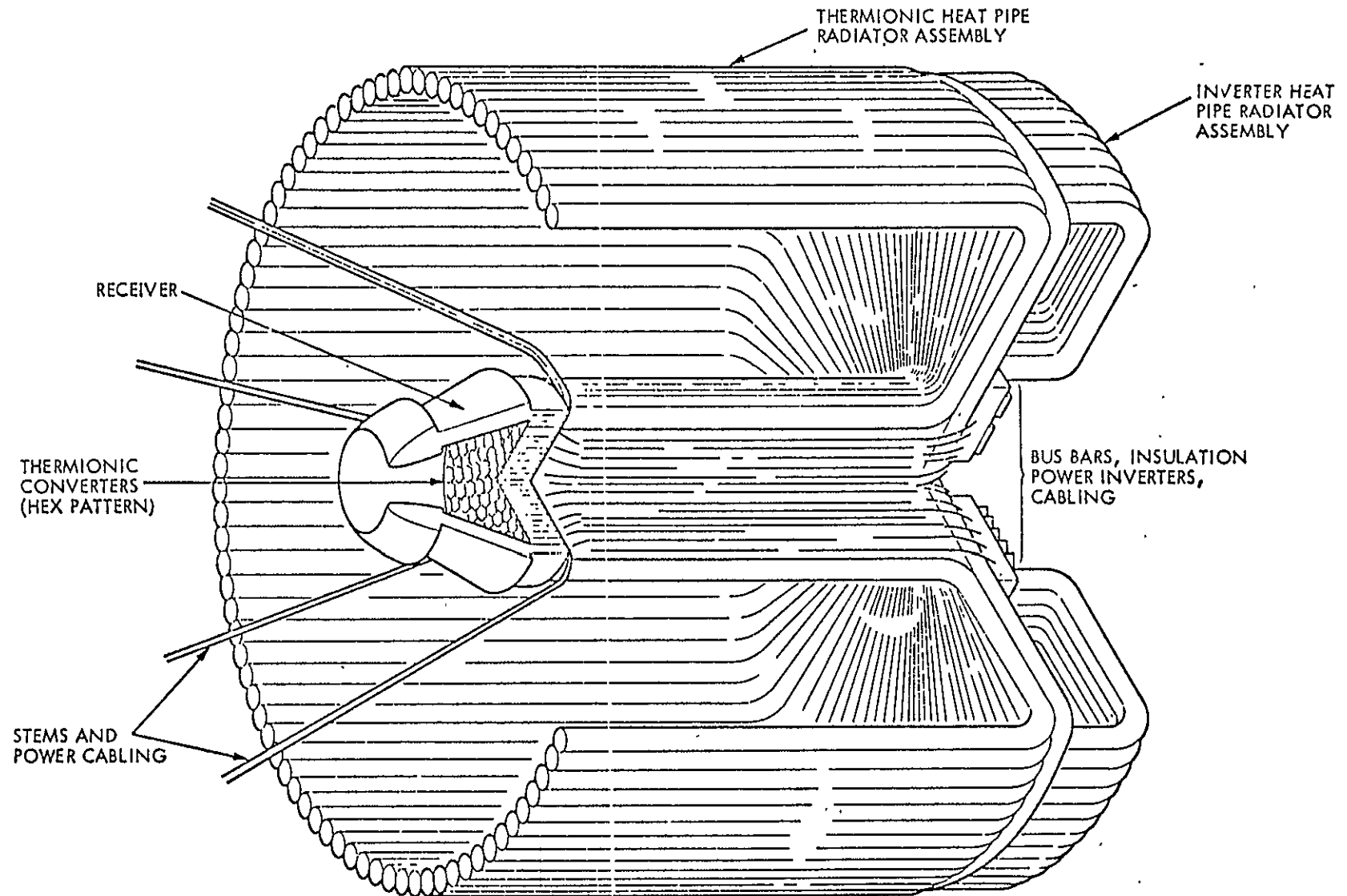


Figure 13. Solar Receiver, Thermionic Converters, Power Processing, and Radiator Assemblies

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ADDENDUM 1

## SOLAR-THERMIONIC SPACE POWER CONCEPTUALIZATION

### 50 kWe System Modifications for Energy Storage

There are several different concepts of renewable energy storage available that may be of some value for use with solar thermionic space power. Among the most interesting are:

- Electrochemical (Battery or Fuel Cell)
- Thermal (Heat of fusion)
- Mechanical (Flywheel)

Each of these energy storage systems is expected to operate at 80-90 percent efficiency. With strong development programs they may ultimately be able to achieve the order of 100 W-H/kg. Such a goal, however, may not be achievable before 1982, when technology is expected to move into flight programs. If we can believe the research literature, the flywheel technology of the future may eventually be able to achieve several hundred W-H/kg (Ref. 1). This appears to be an excellent concept for use with rotating machinery, such as the solar-Brayton system. But even thermionics, where energy is transported electrically, would be suitable with this energy storage by using an electric motor.

Thermal energy storage has a basic simplicity that is most attractive for static energy-conversion, such as the solar-thermionic system. Further study of this concept might be fruitful. An initial survey of materials shows several are available for operating temperatures in the 1400-1500K

1. Post, R.F., and Post, S.F., "Flywheels", pp. 17-23, Scientific American, Vol. 229, No. 6, December 1973.

range. But significant effort is going to be needed in the determination of suitable methods of containment, particularly for the long lifetimes indicated and the large number of thermal cycles involved. Materials limitations are much more significant as we move into the temperature regime where refractories are required for containment. Since there has apparently been no previous work done at high temperature, feasibility must be demonstrated before this concept can be included in system planning.

For the moment, electrochemical storage will be considered as the only energy storage concept where we have good assurance of availability of near-term technology. In this area we will also limit ourselves to battery storage. Although fuel cells are available, efficient electrolysis cells are not developed for space use. Evaluation of potential development here would also be desirable, along with further evaluation of thermal and mechanical energy storage systems.

The NiCd battery development program, now on-going, will provide a rechargeable battery technology by CY 1980 at 44 W-H/kg. At 100% depth of discharge the technology provides 55 W-H/kg, but such performance is incompatible with 5-10 years life in low Earth orbit. Voltage characteristics also indicate that about 85% of the electrical input energy is available as output at the load unless more-sophisticated systems components are added. Input energy needed is therefore 117.6% of output energy.

The energy storage requirements also affect the size of the power conversion system. Since stored energy is required for approximately one-third of the time in low Earth orbit, the power conversion system must be increased by approximately 60% to allow for battery recharging. Thus a nominal 50 kWe system must operate at 80 kWe. With a further requirement for 20% redundancy previously discussed, the system should be sized for 96 kWe, or nearly twice that of minimum sunlit specification.

A 96 kWe peak (BOL) solar-thermionic system will require a solar concentrator of approximately 27 m diameter, with a mass of 2000 kg. This is based on a solar insolation of  $1300 \text{ W/m}^2$ , a 15% efficient concentrator and converter, 10% electrical system losses and 5% inverter losses.

No funds are available for system configuration studies, but approximation is possible by superimposing the first and the last figure of the previous paper (figures attached). Batteries may be mounted to the spacecraft at the point of use or at the solar concentrator. Some thought should be given to optimization of structure, cabling and power inverter mass in the final system design. A summary mass tabulation for a 50 kWe system complete with energy storage is shown in Table 1. A schedule summary is shown in Table 2, and cost summary is in Table 3.

Table 1. 50 kWe Solar-Thermionic System Mass

| <u>Component</u>            | <u>Mass Estimate</u> | <u>Specific Mass</u> |
|-----------------------------|----------------------|----------------------|
| Solar Concentrator Assembly | 2000 kg              | 40 kg/kWe            |
| Battery Assembly            | 680 kg               | 13.6 kg/kWe          |
| Receiver                    | 100 kg               | 2 kg/kWe             |
| Thermionic Converters       | 160 kg               | 3.2 kg/kWe           |
| Bus Bars                    | 100 kg               | 2 kg/kWe             |
| Power Inverter Assembly     | 300 kg               | 6 kg/kWe             |
| Primary Radiator            | 100 kg               | 2 kg/kWe             |
| <u>Cabling</u>              | <u>100 kg</u>        | <u>2 kg/kWe</u>      |
| Total Power Subsystem       | 3540 kg              | 70.8 kg/kWe          |

Table 2. 50 kWe Summary Development Schedule

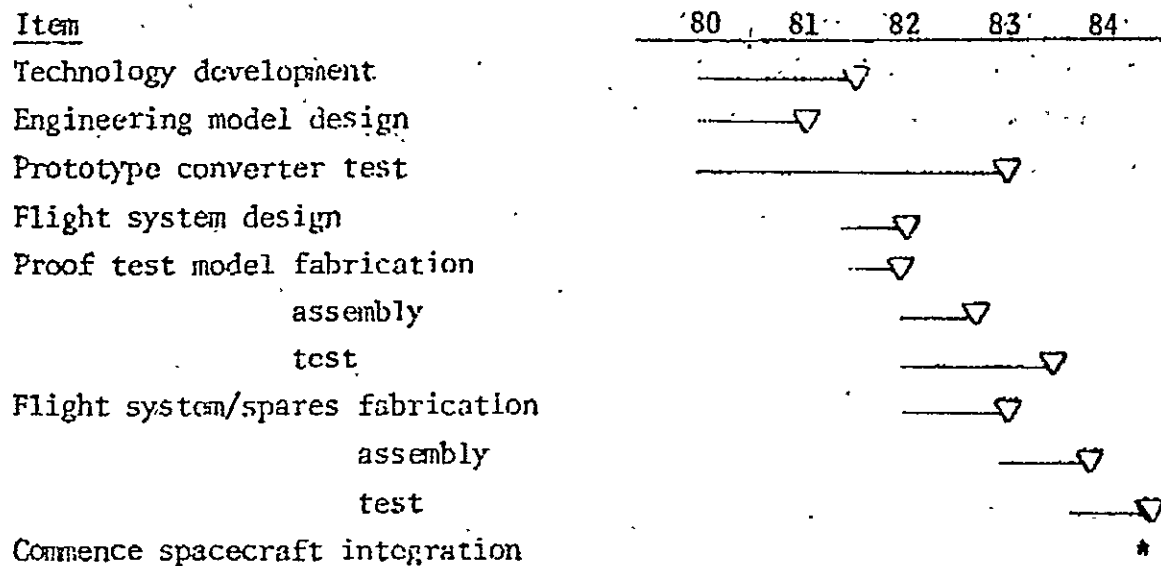


Table 3. 50 kWe System Cost Summary

|                             | <u>80</u>   | <u>81</u>   | <u>82</u>  | <u>83</u>  | <u>84</u>  | <u>Followon Unit<br/>Recurring Cost</u> |
|-----------------------------|-------------|-------------|------------|------------|------------|-----------------------------------------|
| Solar Concentrator Assembly | 0.5         | 0.5         | 6.2        | 8.0        | 0.3        | 7.0                                     |
| Battery Assembly            | 0.25        | 0.25        | 0.5        | 0.8        | 0.2        | 0.7                                     |
| Receiver                    | 0.05        | 0.05        | 0.05       | 0.05       | ---        | 0.001                                   |
| Thermionic Converters       | 2.0         | 2.0         | 4.0        | 6.0        | 3.0        | 0.2                                     |
| Bus Bars                    | 0.1         | 0.1         | 0.3        | 0.3        | 0.1        | 0.05                                    |
| Power Inverter Assembly     | 0.2         | 0.2         | 0.5        | 1.1        | 0.4        | 0.1                                     |
| Primary Radiator            | 0.2         | 0.2         | 0.5        | 1.2        | 0.3        | 0.1                                     |
| <u>Cabling</u>              | <u>0.05</u> | <u>0.05</u> | <u>0.2</u> | <u>0.5</u> | <u>0.2</u> | <u>0.2</u>                              |
| Total Power Subsystem       | 3.35        | 3.35        | 12.25      | 17.95      | 4.50       | 8.35                                    |

### CONFIGURATION

DISH DIAMETER = 27 meters

RADIATOR DIAMETER = 3 meters

RADIATOR LENGTH = 3 meters

### RADIATOR DESIGN

DIMENSIONS ABOVE

TEMPERATURE  $\approx$  750°K

STAINLESS STEEL HEAT PIPES

MERCURY FLUID

FOR 96 KW<sub>e</sub> DESIGN, RADIATOR REQUIREMENT = 480 KW<sub>+</sub>

## REACTOR ELECTRIC POWER FOR SPACE CONSTRUCTION BASE

### I. INTRODUCTION

Currently, a study is underway at LASL to determine the characteristics of various reactor power plants for space applications in order to select a configuration for future ground demonstration and flight. NASA's requirements for a power plant can strongly influence the selection of future reactors for space. At this time, various fuels, reactor designs, shields, converters, and radiators are being considered. The study effort is concentrated on a high temperature, compact, fast reactor that can be coupled with various radiation shielding systems as dictated by the specific mission requirements, and one of several electrical power conversion systems (thermoelectric, thermionics, or dynamic) again depending on mission requirements. It is not necessary to select between these various possible configurations in the evaluation of the relative merits of solar arrays, radioisotopes and reactors as a power source for the space construction base (SCB). Reactor systems can be compared as a class, recognizing that the system of choice will depend on specific requirements for power level, schedule, and the results of further study and comparison of options.

### II. REACTOR TECHNOLOGY

#### A. Mass

The mass of the major components for a reactor power plant are given below.

1. Reactor. It is proposed that a single reactor subassembly be designed for the range of electric power from 50-150 kW(e). The rating of the reactor would be 1000 kW(t), with the reactor run at less than the design power to meet the lower electrical requirements.

The reactor core contains 91 hexagonal fuel elements consisting of 90% UC 10% ZrC (see Fig. 1). Each fuel element measures 24.8 mm across the hexagonal flats and is 250 mm long. The fuel elements are contained in a thin-wall molybdenum can. Each fuel element is cooled by a centrally located, molybdenum heat pipe. The heat pipe is an efficient means of transporting heat from the core. It is a self-contained structure that achieves very high thermal conductances by means of two-phase flow with capillary circulation. The heat transfer is achieved within the contained envelope of the heat pipe by evaporating a liquid, transporting the vapor to another part of the container, condensing it, and returning the condensate to the evaporator through a wick of suitable capillary structure. The fuel is segmented to allow for swelling, to minimize fabrication problems, to prevent



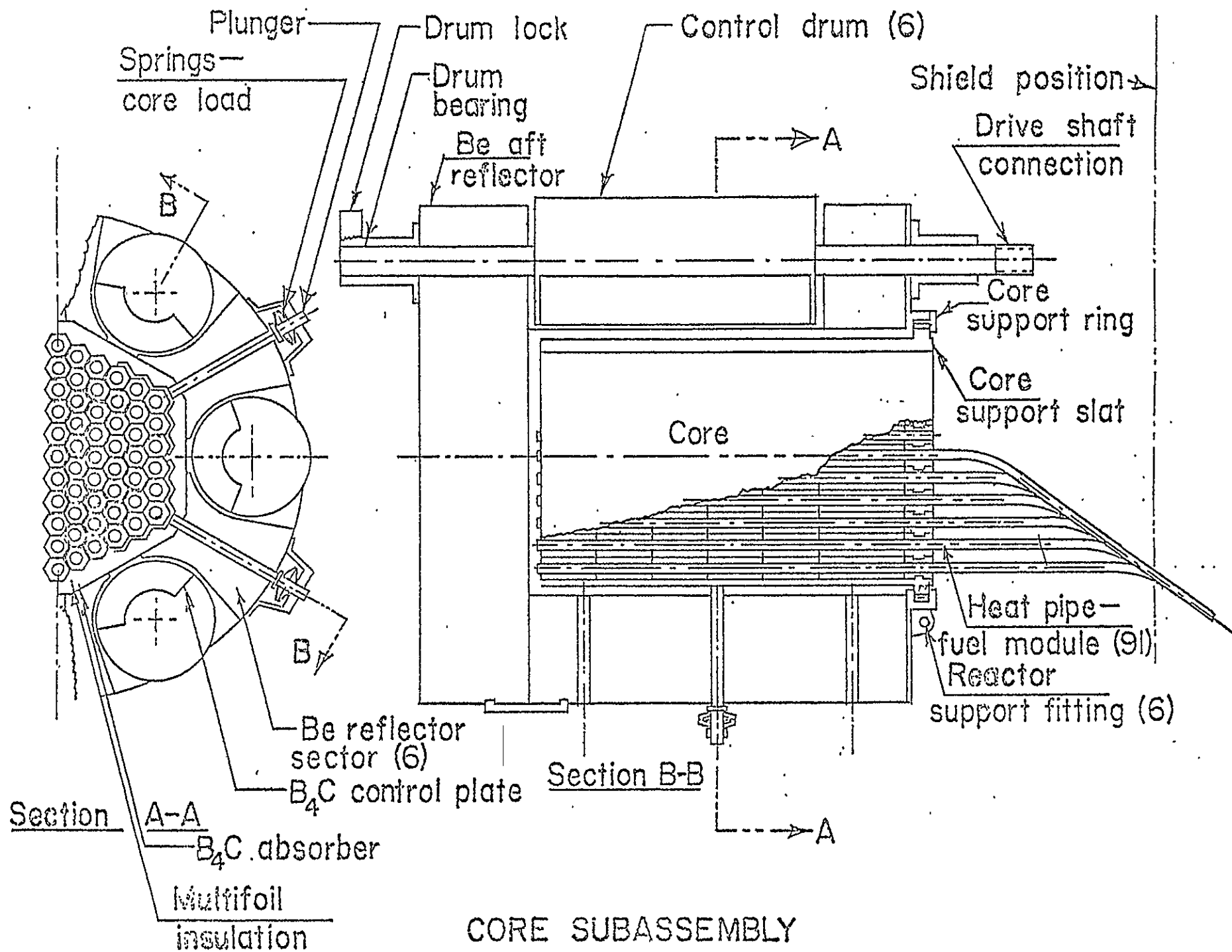


FIGURE 1

bowing and enhance heat transfer to permit variations in uranium loading, and to allow for thermal expansion. Cladding of fuel elements is used for storage and for protection against oxidation.

The core is enclosed in a structure and maintained in compression by a series of spring loaded compression plungers. Multifoil insulation is used to minimize heat transfer from the core to the reflector. The core, with its 91 heat pipes, provides essentially 91 independent loops for removing heat. The loss of a heat pipe will cause slightly elevated temperature in the surrounding heat pipes, but the surrounding elements can easily tolerate the change. The core could thus have several failures that would be considered catastrophic in most reactors but which would cause no major degradation of performance in this reactor design.

The core is surrounded by a neutron reflector of beryllium on the sides and aft end and beryllium oxide at the forward end. Beryllium oxide is required at the heat pipe penetration end because of higher operating temperatures there. The reflector is 100 mm thick. Control of the reactor is provided by changing the position of neutron absorbing material located within the reflector. Both sliding blocks and rotating drums are considered for reactivity control, but, for the reference design, rotating drums were selected because of extensive previous experience. Six drums are used in the reflector, each containing a boron-carbide sector that is rotated for neutronic control. The control surfaces are rotated in discrete steps by an actuator that is placed behind the shield to reduce the incident nuclear and thermal radiation. The reactor power will be controlled to maintain a constant outlet voltage from the power conversion units in such a way as to minimize thermal cycling of the reactor.

The mass of the core is 325 kg for thermoelectric or dynamic converters. If thermionics is selected, higher temperatures are required. The fuel would be uranium dioxide rather than the UC-ZrC and the resulting mass of the core increased to about 545 kg for 1000 kW(t).

2. Shield. The shield mass depends on power level, location and biological requirements. Locations considered are attachment to the space construction base by a short boom (10 m), a long boom (100 m), a tethered cable (1000 m), or a separate satellite where the electric energy is microwaved back to the SCB (5000 m). For the boom configurations it is considered acceptable to have different radiation levels at the aft end of the power plant away from the SCB because of the rigid arrangement. The tethered arrangement and separate power satellite may need equal shielding on all sides of the reactor ( $4\pi$ ) because of orientation uncertainties.

Table I provides a list of requirements as defined by NASA in the Space Transportation Payload Safety Guidelines Handbook. NASA recommends a safety factor of 100 on top of these requirements. The duty cycle for SCB workers needs definition in establishing radiation specifications in the vicinity of the reactor.

Sufficient studies have not been performed on manned shielding to provide optimized designs for each configuration. However, information was available from a study by Oak Ridge National Laboratory reported in "Optimization of a Shield for a Heat-Pipe Cooled Fast Reactor Designed as a Nuclear Electric Space Power Plant," ORNL-YM-3449, June 15, 1971 to provide a good base for initial power plant studies. The study considered two core angles - 45 and 90° for the shield. If the power plant is located in the middle of one side of the SCB, the short boom (10 m) would require a 90° shield while with the long boom a 45° shield should be adequate. If the power plant is located at one end of the SCB, a 45° shield (or less) would suffice for either boom. Pictures of the shield configuration are shown in Figs. 2 and 3. Air reactor is slightly different with the same diameter core but slightly less height. Our core has more reflector but this also acts as shielding. Therefore, for the thermoelectrics and dynamic systems no adjustment is made for differences in reactor dimensions. For therionics, a larger core is necessary.

Figure 4 shows the thickness of W and LiH layers for a two-cycle shield as a function of dose rate. Thirty-five rem/quarter from Table I corresponds to 16 m rem/h. The use of 3 m rem/h as a base seems very reasonable at this time. The effect on outer shield radius can be interpolated from Fig. 4. Radiation attenuation varies as the distance squared and linearly with power level. Moving the 450 kW(t) reactor from 30 m (100 ft) to 100 m would reduce the radiation intensity by a factor of 11. Using Fig. 4, instead of an outer shield radius of 114 cm the radius could be reduced to 97 cm. Since the shield weight of spherical arrangement varies as the cube of the radius, roughly a savings of 39% can be expected in shield mass in moving the reactor from 30 m to 100 m from the SCB. The 45° shield mass is given as 6675 kg. Thus, a 45° shield at 100 m would have a mass of 4100 kg. A 90° shield at 30 m has a mass of 11,620 kg. At 10 m the outer shield radius increases from 112 cm to 126 cm which increases shield mass to 16,400 kg. As the space shuttle can deliver 25,000 kg to 500 km orbit, the shield would be very heavy at 10 m but quite acceptable located at 100 m from the SCB.

TABLE I

RADIATION EXPOSURE LIMITS AND EXPOSURE RATE  
CONSTRAINTS FOR UNIT REFERENCE RISK\*

From Space Transportation System Payload Safety Guidelines Handbook, NASA  
Lyndon B. Johnson Space Center, Houston, TX, July 1976, Table 3.14-III.

| Constraints               | REM**                 |                   |               |
|---------------------------|-----------------------|-------------------|---------------|
|                           | Bone Marrow<br>(5 cm) | Skin<br>(0.01 mm) | Eye<br>(3 cm) |
| 1-year average daily rate | 0.2                   | 0.6               | 0.3           |
| 30-day maximum            | 25                    | 75                | 37            |
| Quarterly maximum         | 35                    | 105               | 52            |
| Yearly maximum            | 75                    | 225               | 112           |
| Career                    | 400                   | 1200              | 600           |

\* For details, see "Radiation Protection Guides and Constraints for Space Missions and Vehicle Design Studies Involving Nuclear Systems, Report of the Radiobiological Advisory Panel of the Committee on Space Medicine, Space Science Board, National Academy of Science, 1970."

\*\* REM (Roentgen equivalent man) REM is a unit of radiation dose equivalent. For details, see "International Commission on Radiation Protection," Publication No. 9, 1966.

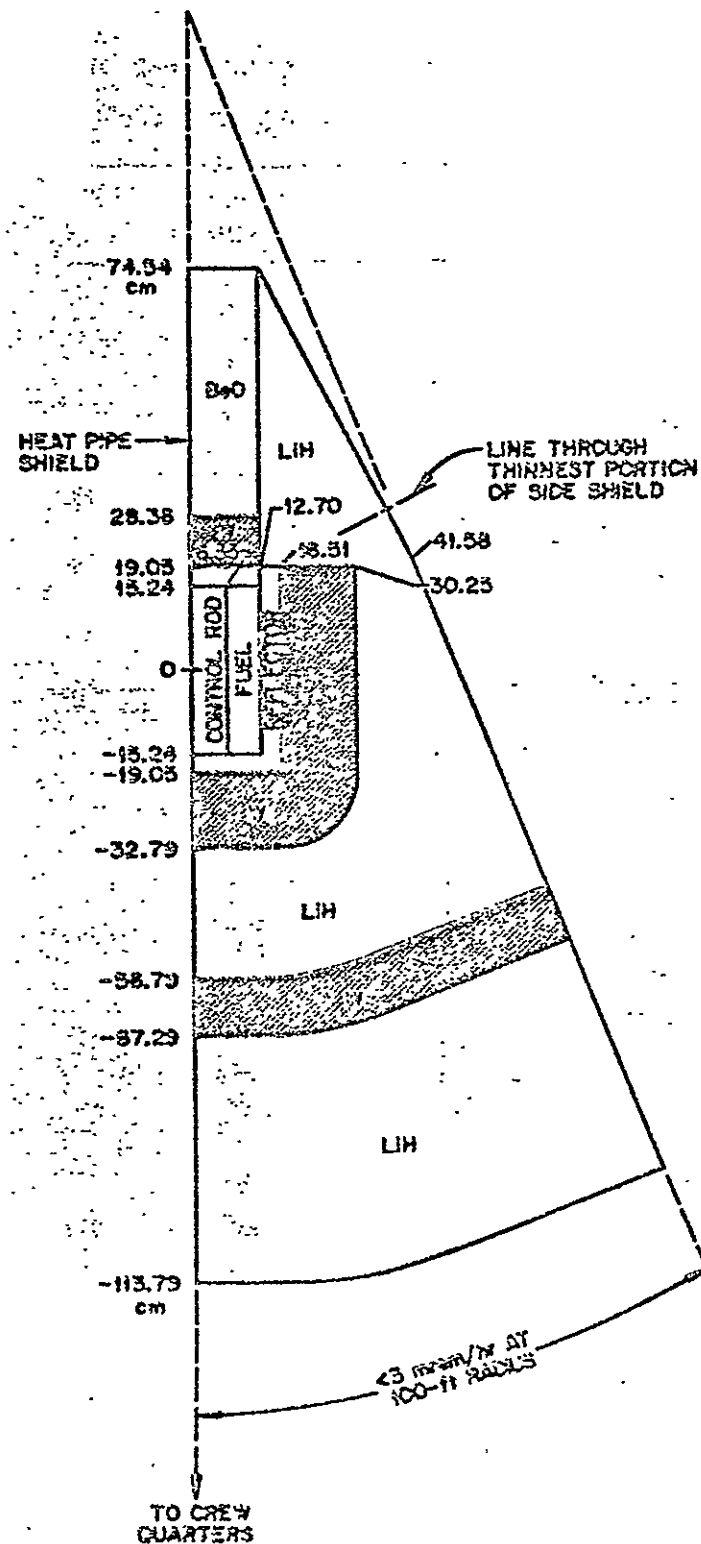


FIGURE 2  
Configuration for a Two-Cycle Asymmetric Shield  
With a 45° Cone Angle (450 kW(t)).

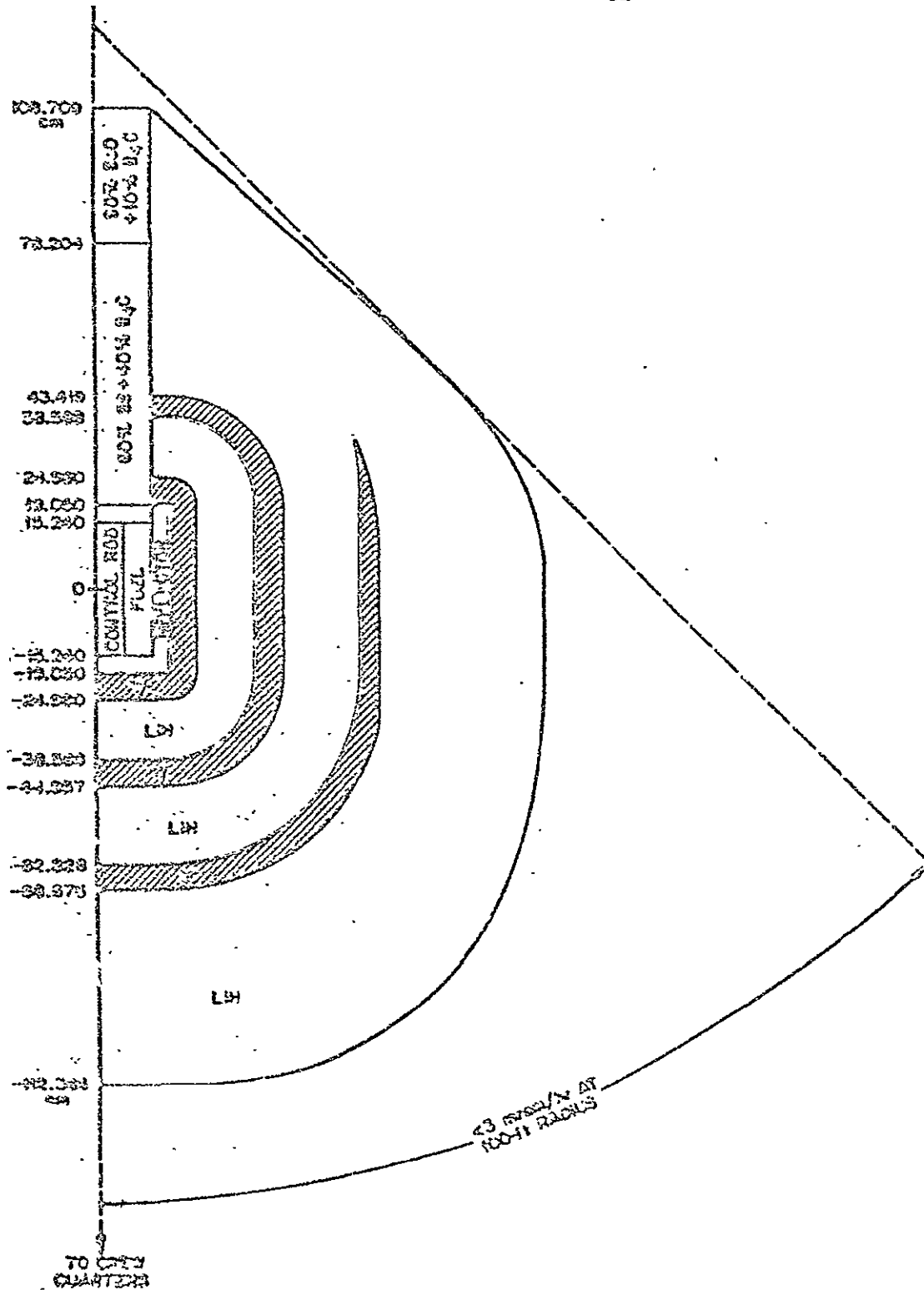


FIGURE 3

Optimized Asymmetric Shield with a 90° Cone Angle (450 kW(t))



For the tethered arrangement, a  $4\pi$  shield of 14,500 kg is used as a base. Moving out to 1000 m would attenuate the radiation by a factor of 1100 or only 72 cm outer radius is needed for a shield mass of 3630 kg. At 5000 m, the additional geometric attenuation is  $27 \times 10^3$ . The outer radius decreases to 67 cm and mass to 2970 kg.

For thermionics, the larger core would increase the shield diameter. This would result in a 25-30% increase in the shield mass. Table II summarizes the preliminary shield mass numbers. Typical isodose plots are shown in Figs. 5 through 8. Table III provides mass figures as a function of distance, power level and shield configuration. Figure 9 is a plot of Table III.

3. Converters. The converter selection depends on power level, specific missions requirements, trade-offs between size and weight, judgment on the amount of redundancy desired, and the state-of-development of the converter systems.

The electrical power from thermoelectric modules (T/E) depends on operating temperatures and scale linearly with the number of T/E modules. The specific mass of the modules is 6.8 kg/kW(e). This includes provisions for a common T/E converter heat sink ring to thermally cross couple the T/E modules with redundant heat rejection paths.

Dynamic converter specific mass depends on the number of redundant loops desired. Representative values are given in Fig. 10.

Thermionic systems like the thermoelectrics also tend to vary as a linear function with power level. Specific mass is 2.1 kg/kW(e) for the near term and projection on advanced designs indicate values as low as 0.8 kg/kW(e). It should be noted that thermionic systems required higher temperatures which impact on reactor weight.

4. Radiator. The heat rejection requirements depend on such factors as the converter efficiency, electric power level, need for thermal energy in other parts of the space station, space station thermal environment, and maintainability. The radiator area for end-of-life can be computed from the equation:

$$Q_{\text{rejected}} (w) = \text{Area (m}^2) \times 5.67 \times 10^{-8} \left( \frac{w}{\text{m}^2 \text{ K}^4} \right) (0.9) \left( T_{\text{radiator}}^4 (\text{K}) - T_{\text{space}}^4 (\text{K}) \right)$$

where  $Q_{\text{rejection}} (w) = w(e) \left( \frac{1 - \eta}{\eta} \right)$   $\eta$  being the system efficiency.



TABLE II  
SHIELDING MASS

| Location            | Attached   |        |           | Tethered Cable | Power Satellite<br>Microwaving Electricity |
|---------------------|------------|--------|-----------|----------------|--------------------------------------------|
|                     | Short Boom |        | Long Boom |                |                                            |
|                     | End        | Middle |           |                |                                            |
| Distance, m         | 10         | 10     | 100       | 1000           | 5000                                       |
| Thermal Power, kW   | 450        | 450    | 450       | 450            | 450                                        |
| Shield Thickness, m | 1.26       | 1.26   | 0.97      | 0.72           | 0.67                                       |
| Type Shield         | 45°        | 90°    | 45°       | 4π             | 4π :                                       |
| Mass, kg            | 9390       | 16,400 | 4100      | 3630           | 2970                                       |



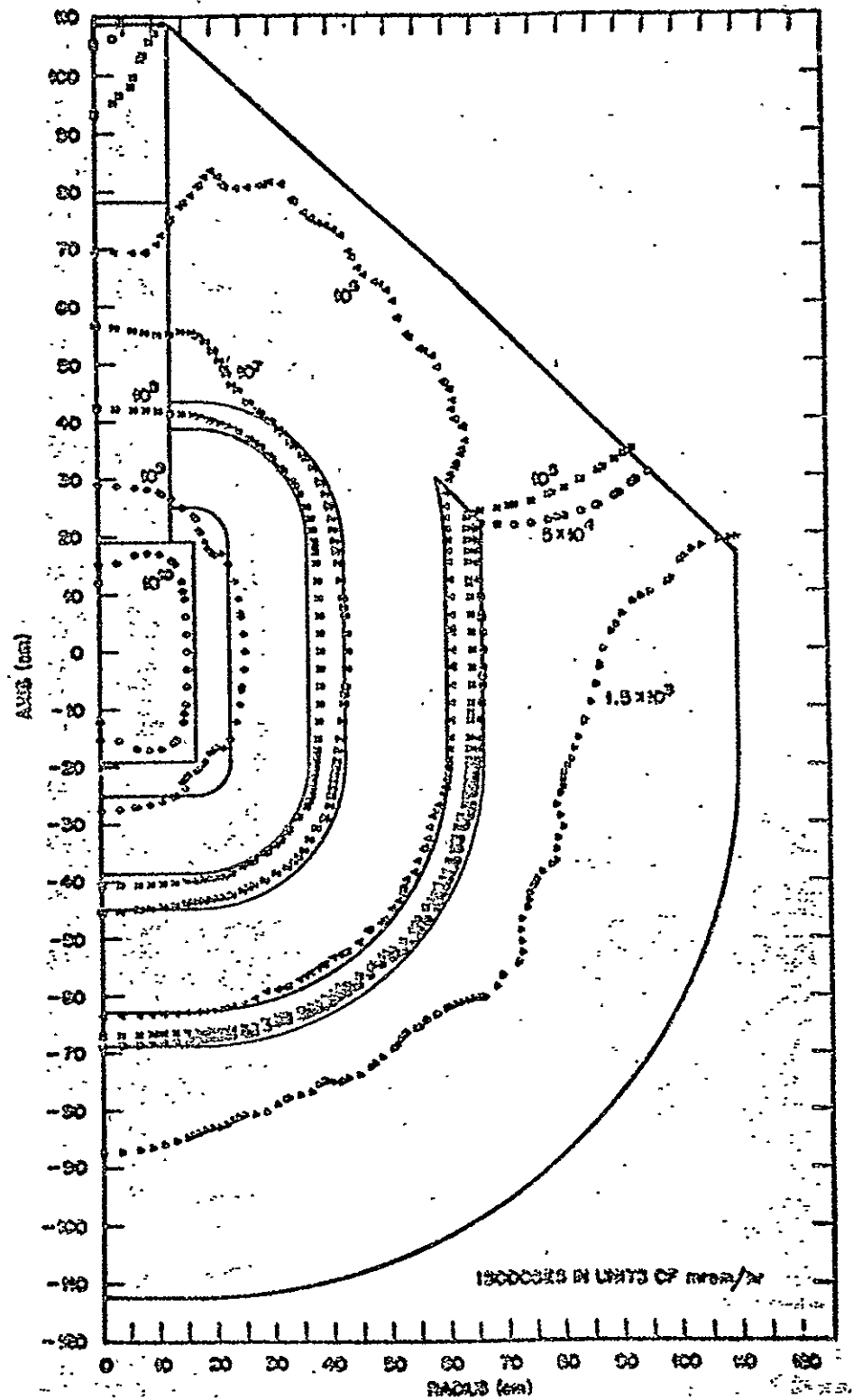


FIGURE 6

Gamma-Ray Isodose Plots for Preliminary Asymmetric Shield with a  $90^\circ$  Cone Angle.

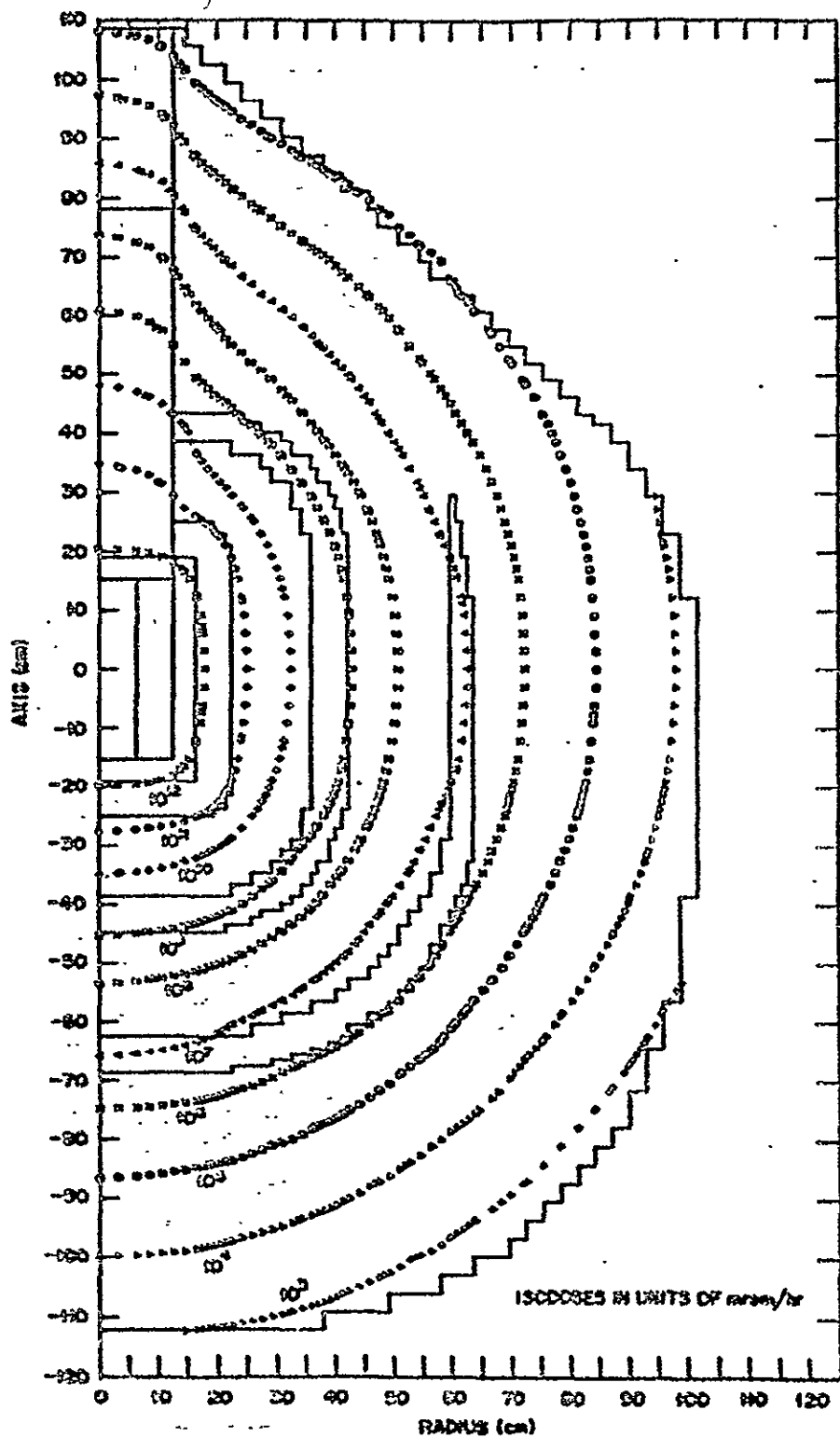


FIGURE 7  
Neutron Isodose Plots for Optimized Asymmetric Shield  
With a 90° Cone Angle.

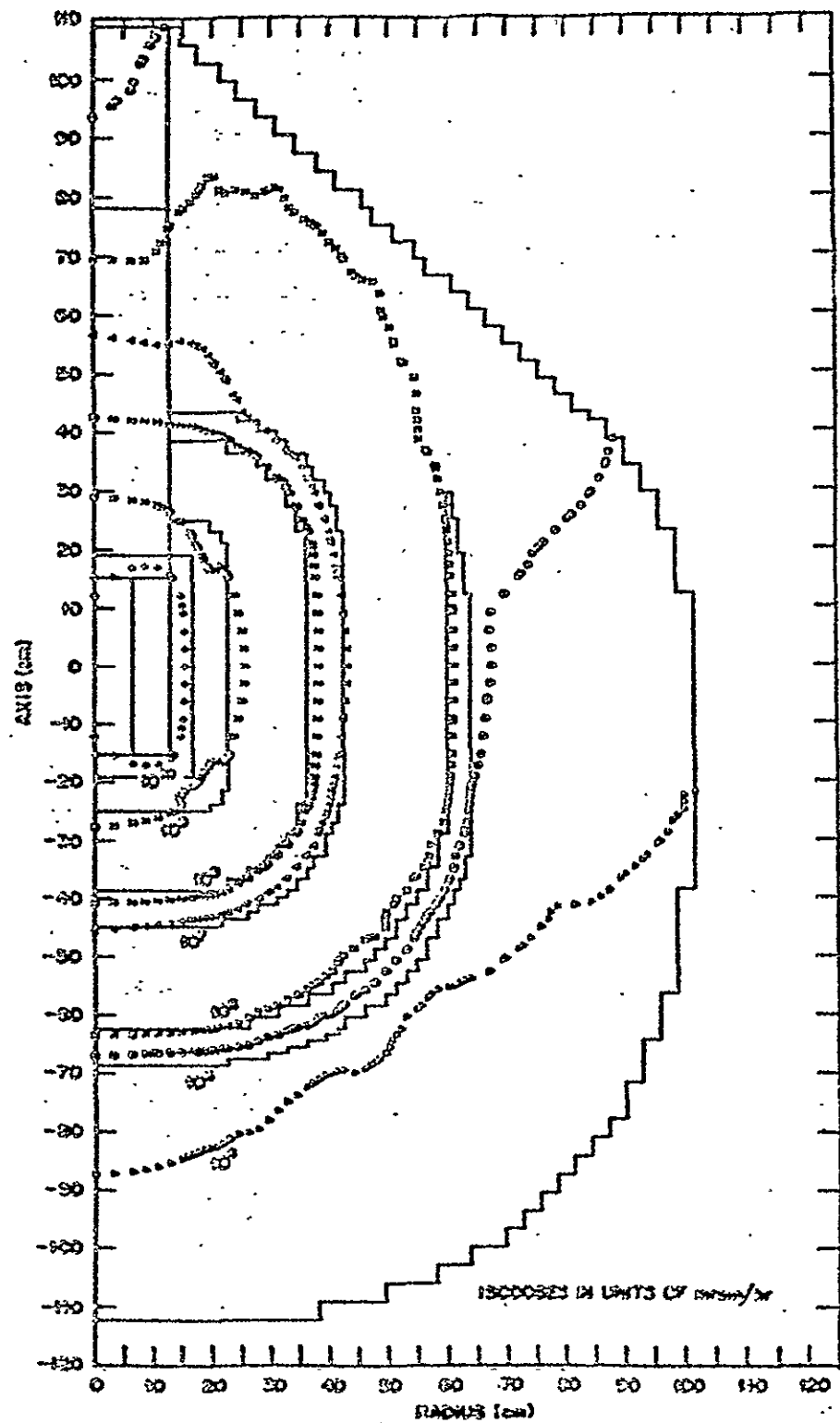


FIGURE 8

Gamma-Ray Isodose Plots for Optimized Asymmetric Shield  
With a 90° Cone Angle.

TABLE III  
MANNED SHIELD MASS, kg

| Distance, m | 45° Shield     |        |        |
|-------------|----------------|--------|--------|
|             | Power, kW(t)   |        |        |
|             | 200            | 600    | 1000   |
| 10          | 8,590          | 11,330 | 11,840 |
| 100         | 3,620          | 4,370  | 4,510  |
| 1000        | 930            | 1,750  | 1,900  |
| 5000        | 560            | 830    | 930    |
|             |                |        |        |
|             | 90° Shield     |        |        |
|             |                |        |        |
|             |                |        |        |
| 10          | 14,950         | 19,730 | 20,610 |
| 100         | 6,310          | 7,610  | 7,840  |
| 1000        | 1,610          | 3,050  | 3,310  |
| 5000        | 980            | 1,450  | 1,610  |
|             |                |        |        |
|             | 4 $\pi$ Shield |        |        |
|             |                |        |        |
|             |                |        |        |
| 10          | 18,670         | 24,620 | 25,720 |
| 100         | 7,870          | 9,500  | 9,790  |
| 1000        | 2,010          | 3,810  | 4,130  |
| 5000        | 1,220          | 1,810  | 2,010  |

Fig. 9

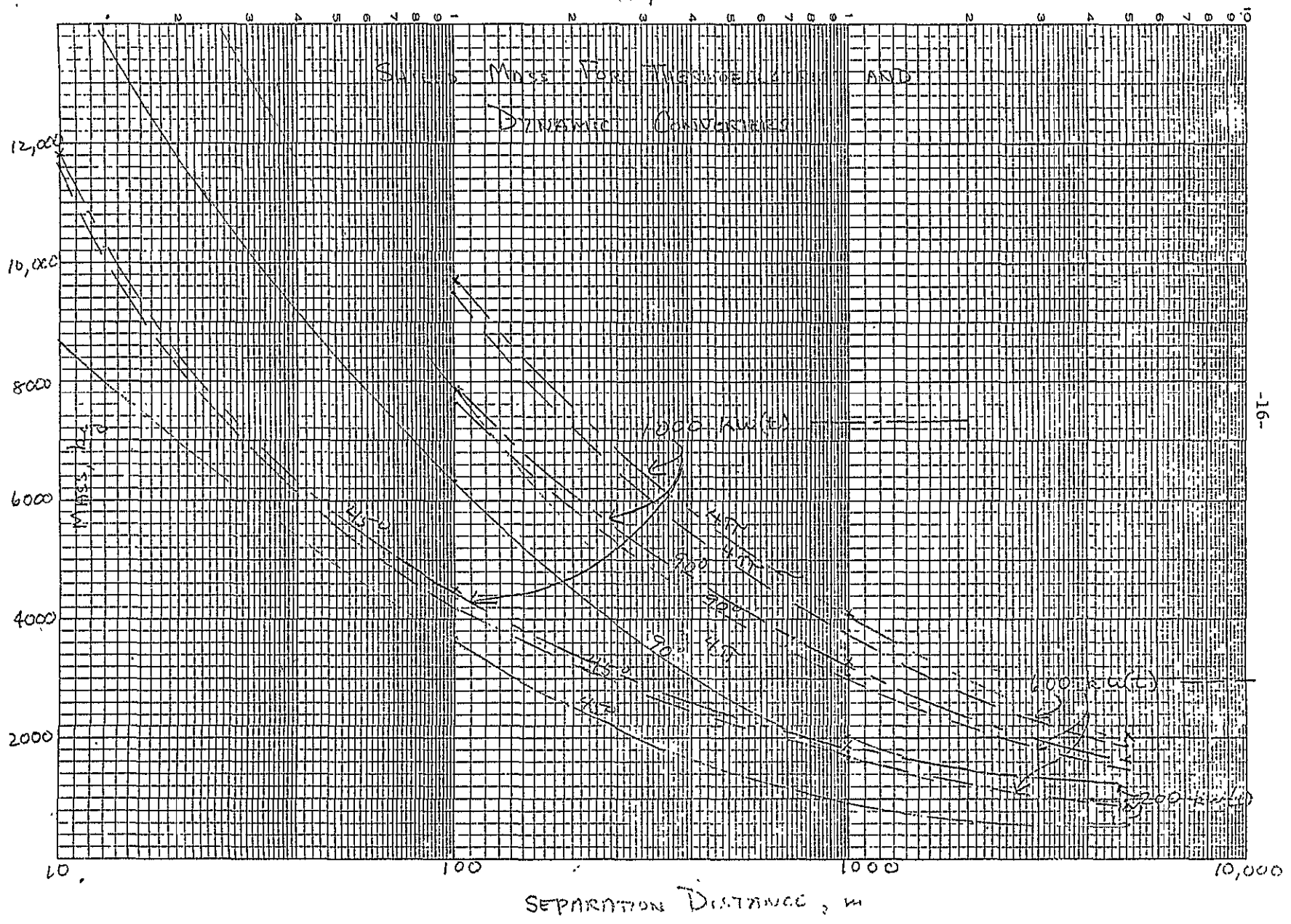


FIG. 10

# BRAYTON CONVERTER MASS





For 500 km orbit,  $T_{\text{space}} = 190\text{-}240$  K depending on space station position relative to the earth and sun. Currently, thermoelectrics are being designed for 775 K radiator temperatures, Brayton for 475 K, and thermionics for 925 K. Current designs are based on using heat pipes. Radiator mass as a function of power is plotted in Fig. 11.

5. Summary. Table IV summarizes the mass for various configurations. The shield will need to be varied for different possible locations and specifications. Power transmission and attachment equipment are not included in the mass calculations.

## B. Operating Characteristics

1. Dimensions. The reactor is a right circular cylinder whose dimensions for the thermoelectric and dynamic systems are 450 mm diameter and 450 mm height. For thermionic systems the core diameter is increased to 550 mm. The converter dimensions are shown in Table V. Packaging of the converters are flexible, for instance, the converters can be packaged inside the radiator as shown in Fig. 12.

2. Temperatures. Thermoelectric system temperatures for the reactor heat pipes and the hot side of the converter are 1275 K while the radiator operates at 775 K. Higher efficiencies are obtained with lower radiation temperatures.

Typical dynamic converter operating temperatures are shown in Fig. 13.

The thermionic reactor heat pipes and cold junction operate at 1675 K while the radiator operates at 975 K.

3. Fluids. The reactor heat pipes contain sodium as the working fluid for the thermoelectrics and dynamic converters and lithium for thermionics. The dynamic converter uses a helium-xenon mixture. The radiator fluid depends on the operating temperature with potassium being used for thermoelectrics, Freon a leading candidate for the dynamic cycle and sodium for thermionics.

## 4. Output Power.

a. Thermoelectrics and Thermionics. These are low voltage devices that can be combined in a series parallel networks. The usual output voltage is 29 V.

b. Dynamic Converter. The dynamic converter supplies the most power at 120 vdc, but by adding other power conversion components, any desired voltage and frequency combination may be obtained.

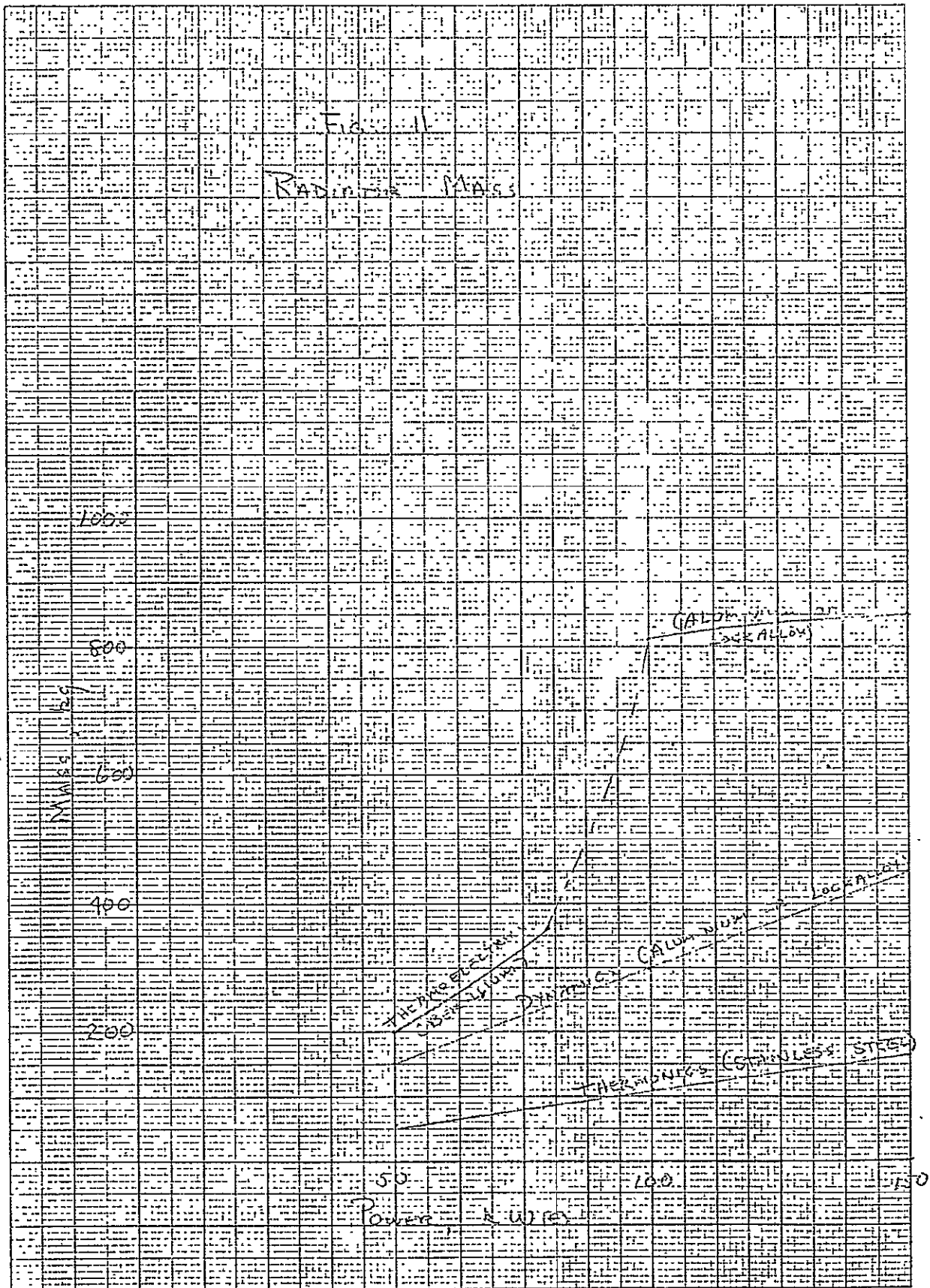


TABLE IV  
MASS PARAMETERS

|                         | 50 kW(t)        |          |             | 100 kW(e)       |          |             | 150 kW(e)       |          |             |
|-------------------------|-----------------|----------|-------------|-----------------|----------|-------------|-----------------|----------|-------------|
|                         | Thermoelectrics | Dynamics | Thermionics | Thermoelectrics | Dynamics | Thermionics | Thermoelectrics | Dynamics | Thermionics |
| Reactor power, kW(t)    | 625             | 200      | 250         | 1000            | 400      | 500         | 1000            | 600      | 750         |
| Efficiency, %           | 8               | 25       | 20          | 10              | 25       | 20          | 15**            | 25       | 20          |
| Radiator power, kW(t)   | 575             | 150      | 200         | 900             | 300      | 400         | 850             | 450      | 600         |
| Radiator Temperature, K | 775             | 475      | 975         | 575             | 475      | 975         | 475             | 475      | 975         |
| <u>Mass (kg)</u>        |                 |          |             |                 |          |             |                 |          |             |
| Core                    | 325             | 325      | 545         | 325             | 325      | 545         | 325             | 325      | 545         |
| Shield*                 | 4380            | 3625     | 4840        | 4505            | 4100     | 5440        | 4505            | 4370     | 5680        |
| Converter               | 340             | 490      | 110         | 680             | 720      | 210         | 1020            | 930      | 315         |
| Radiator                | 200             | 150      | 55          | 810             | 300      | 110         | 850             | 450      | 165         |
| Structure               | 85              | 95       | 70          | 180             | 135      | 90          | 220             | 170      | 100         |
| TOTAL                   | 5330            | 4685     | 5620        | 6500            | 5580     | 6395        | 6920            | 6245     | 6805        |

\* Shield based on 100 m boom located on the side of the space base (45° shield). This should be adjusted for other locations using Table III or Fig. 9.

\*\* Advanced technology projected to be available in mid-1980's.

TABLE V

DIMENSIONS OF CONVERTERSThermoelectricsFor 91 heat pipe reactor

| <u>Power, kW(e)</u> | <u>Length, mm</u> |
|---------------------|-------------------|
| 10                  | 70                |
| 50                  | 350               |
| 100                 | 700               |

| <u>Dynamic Converter (per unit) mm</u> | <u>Power Level, kW(e)</u> |                 |                 |                 |
|----------------------------------------|---------------------------|-----------------|-----------------|-----------------|
|                                        | 12.5                      | 25              | 37.5            | 50              |
| Rotating machinery                     | 305 x 635                 | 305 x 660       | 305 x 686       | 330 x 711       |
| Recuperator                            | 381 x 152 x 508           | 483 x 127 x 483 | 584 x 127 x 483 | 660 x 127 x 508 |
| High temperature heat exchanger        | 350                       | 350             | 350             | 350             |
| Radiator heat exchanger                |                           |                 |                 |                 |

Thermionics

| <u>Power, kW(e)</u> | <u>Length, mm</u> |
|---------------------|-------------------|
| 50                  | 85-150            |
| 150                 | 250-450           |

# REACTOR SUBASSEMBLY WITH BRAYTON CONVERTER

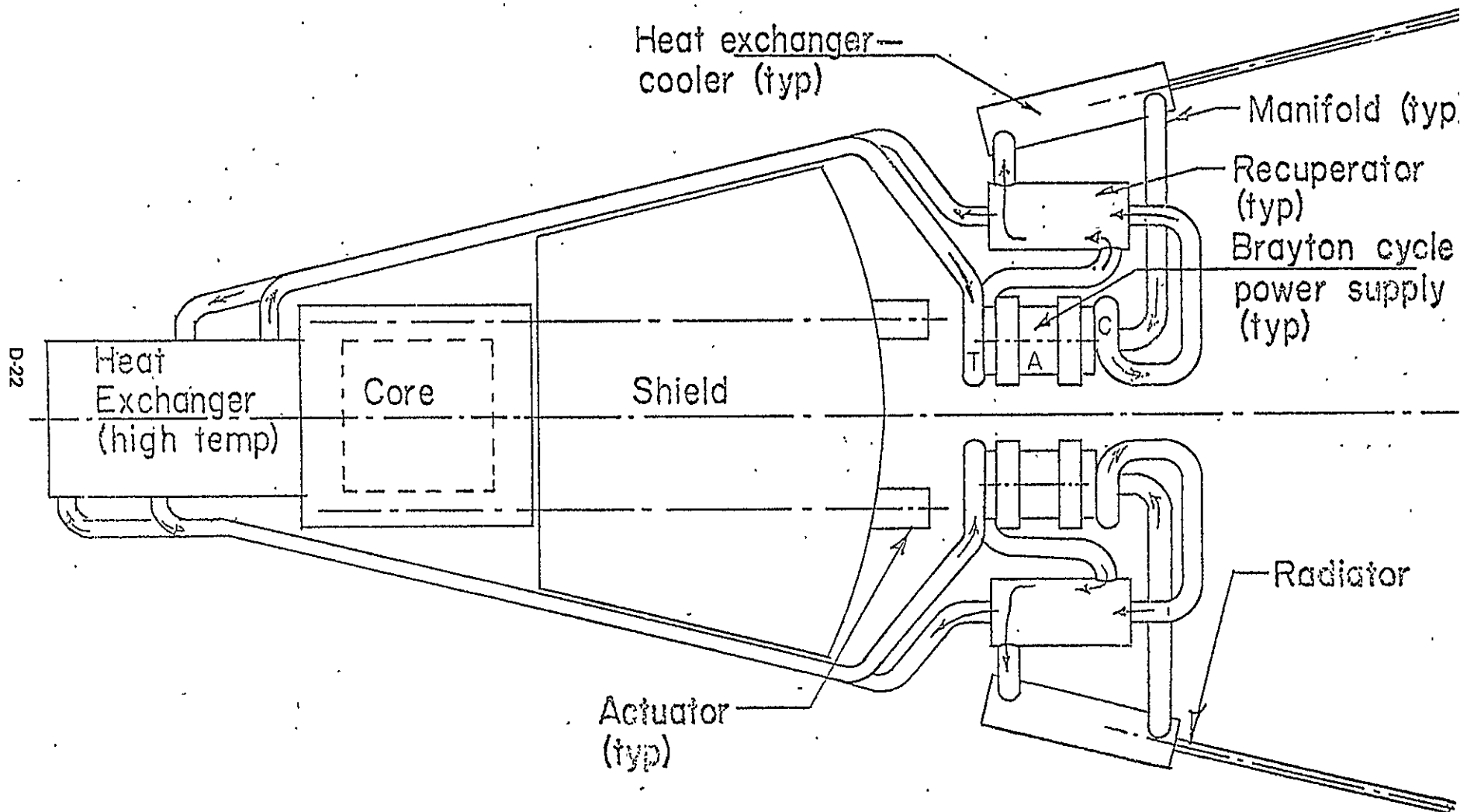


FIGURE 12

# BRAYTON CYCLE POWER SYSTEM

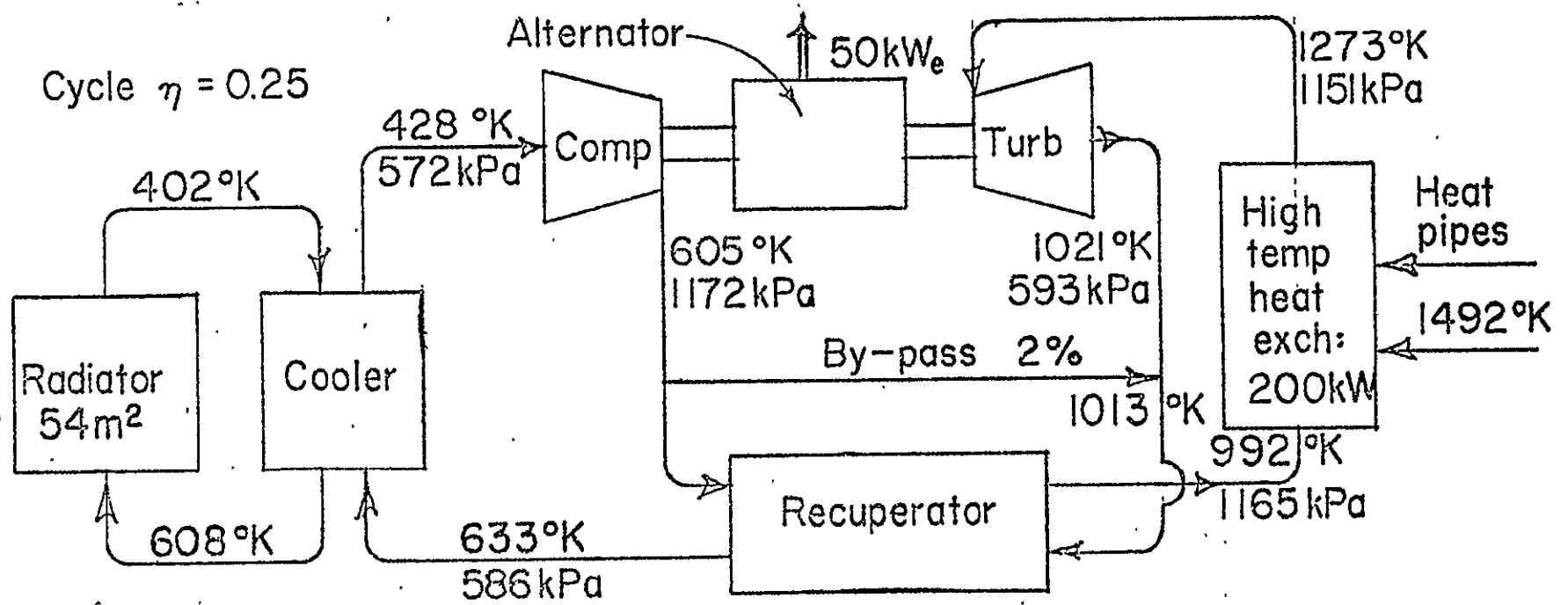
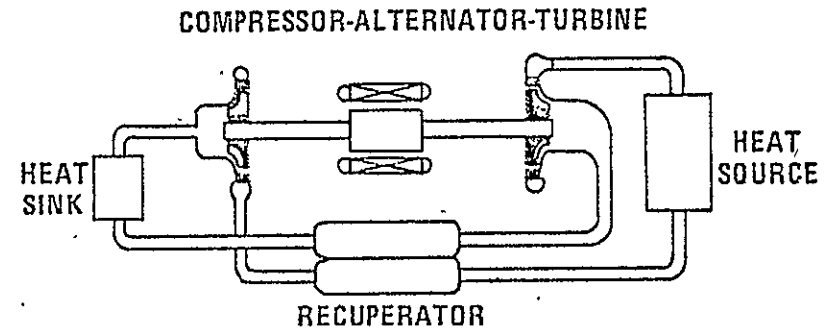
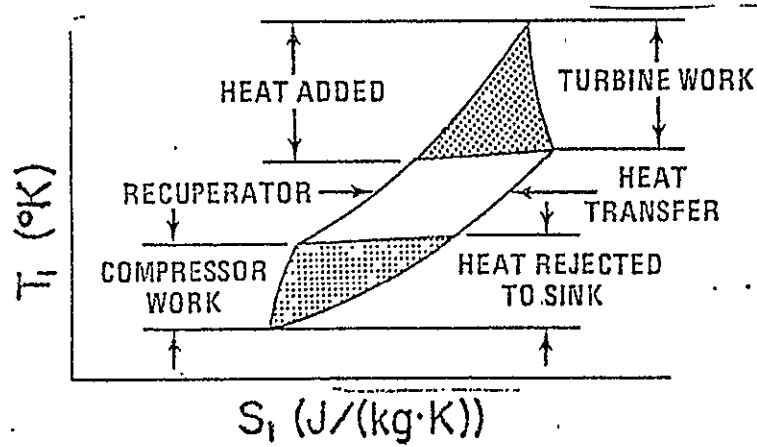


FIGURE 13

### C. Orbiter-Space Transportation Layout

A conceptual layout in the orbiter have been made for the 100 m boom configurations considering volume and mass limitations but no detailed studies that consider center of gravity or structure have been made. Two complete power plants plus a section of the space construction base can probably be packaged in a single shuttle load (Fig. 14). The power plants would be removed from the shuttle module once in orbit and placed in their operating position. The outer surface of the bay could be used as a module for the space station. The control equipment would remain inside the station module.

### D. Power Module Layouts

During the shielding discussion, locations for the power module were discussed. Two modules can be incorporated in the SCB with one operating and the other on standby. In the boom configuration, the standby module can be retracted for added shielding to the SCB. The control consoles and power substation would be the only elements located in the SCB. No gimbaling or orientation mechanism is necessary with the reactor power module.

For the tethered line or separate satellite, the reactor does not need orientation but an orientation mechanism may be required for the tethered line configuration to reduce drag and for the separate satellite for the microwave system.

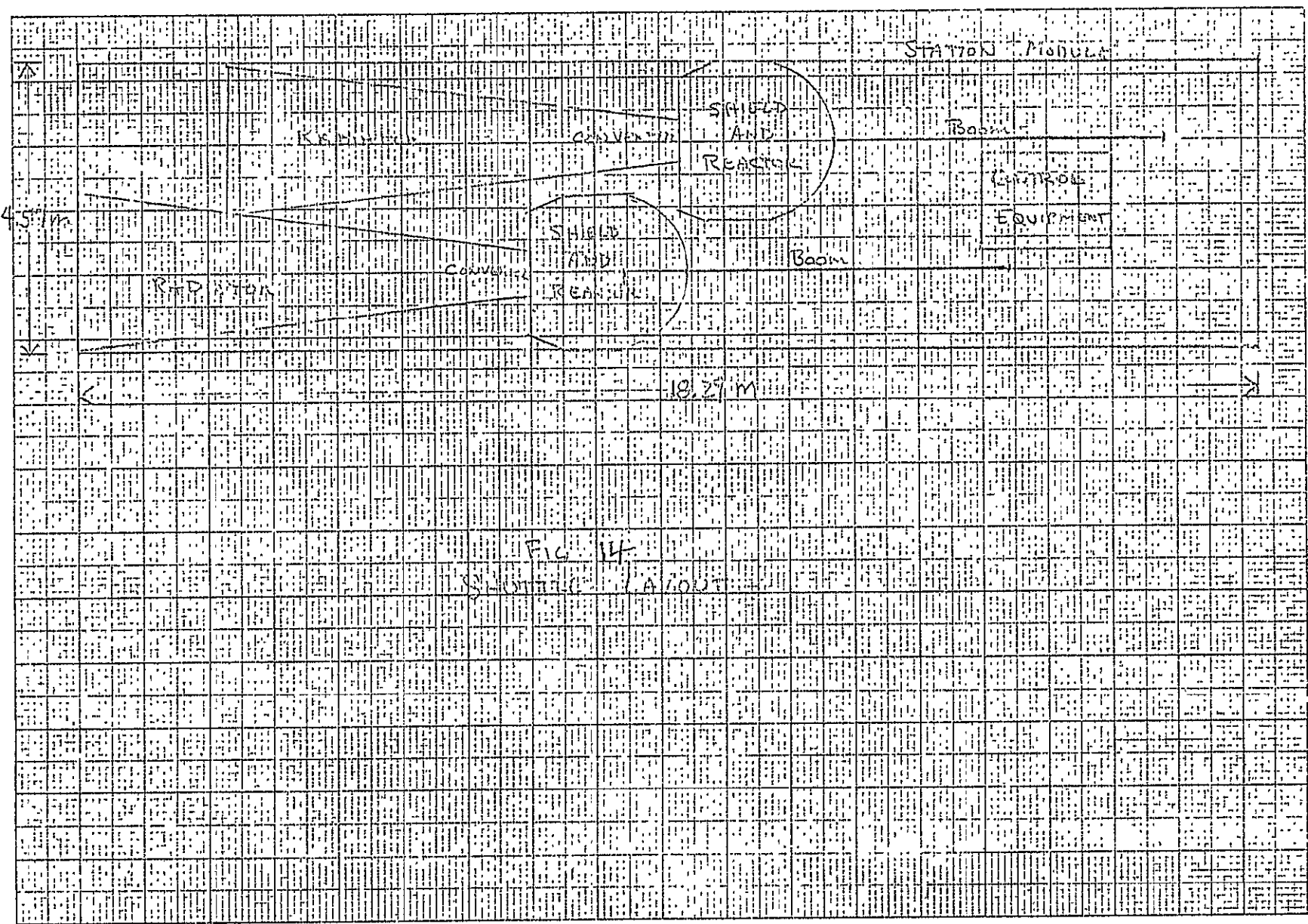
### E. Schedule

The availability of a flight unit depends on the definition of a flight unit and budgets. Facilities are in existence so that they are not a constraining item. Based on adequate funding to start a ground demonstration program in October 1979, a power plant could be put on test in October of 1981 and a flight unit with qualified hardware delivered in October of 1984. This schedule does depend on a better definition of a flight unit. Figure 15 shows the schedule for delivery of the flight unit in 1985.

### F. Costs

1. Development Costs Borne by NASA. The cost of the ground demonstration program and first flight unit are borne by ERDA.

2. Subsequent Flight Units. Flight units beyond the initial one will cost approximately that shown in Fig. 16 (1978 dollars) for manufacturing. These units would be paid for by NASA. Flight support costs depend on definition of the type of support required.





## PROGRAM PLAN

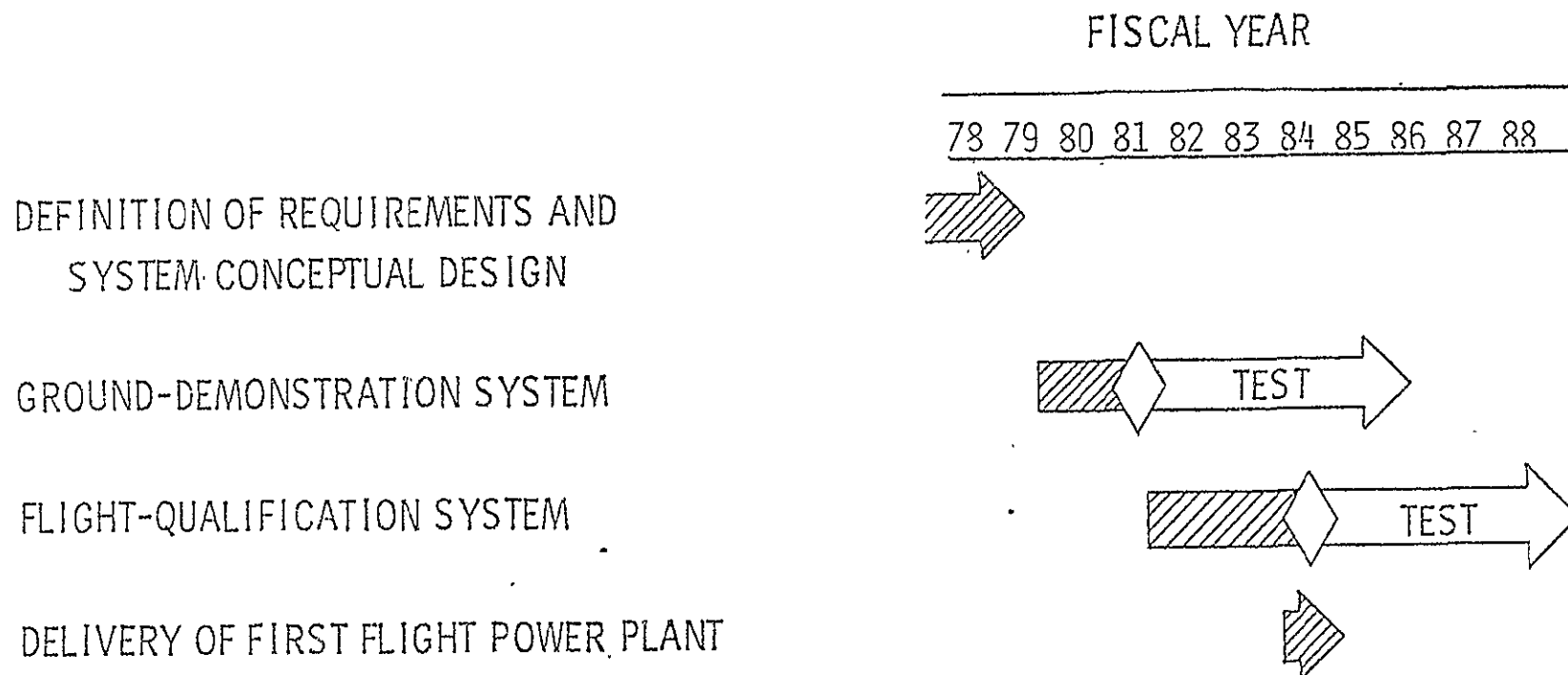
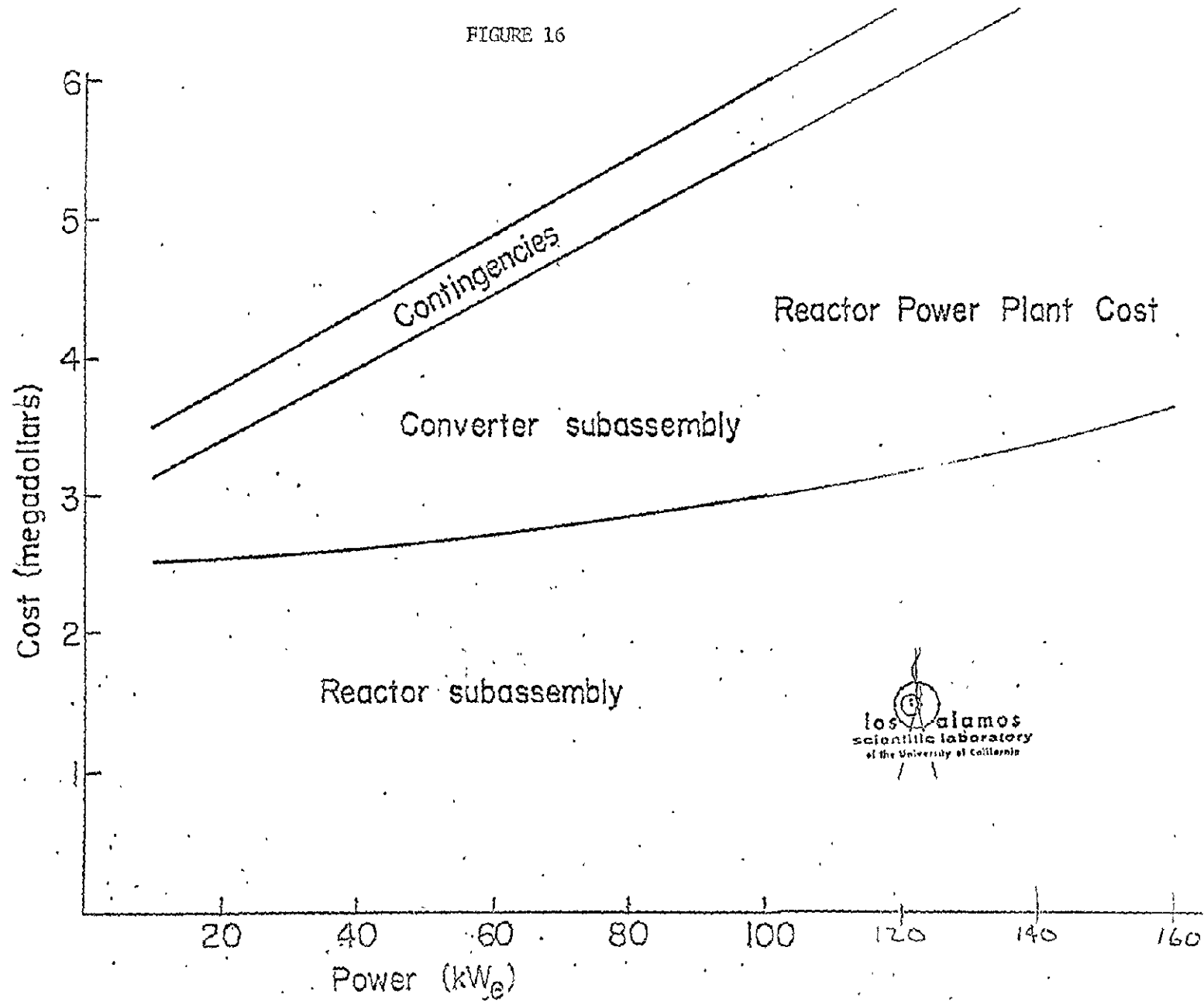


FIGURE 15

FIGURE 16



### 3. Fuel Costs. Included in unit cost.

### G. Current Development Status

Zirconium hydride fueled reactors were under development and actually flight tested (SNAP-10A). However, temperature limitations (975 K) made the system bulky and complex. Higher temperature fuel elements using uranium-carbides, uranium-dioxides, and uranium-nitrides could be further developed in place of zirconium hydride (UH-ZrH) fuels. This permits long term reactor operation at temperatures of 1675 K. The uranium-carbide fuels were extensively developed for the nuclear rocket and thermionic programs and continued for high temperature gas-cooled and liquid metal fast breeder reactors.

An examination of technologies shows the following developments may be applied to power plants for the mid-1980's and beyond:

- Fuel element materials including uranium-carbides, uranium carbide-zirconium carbide, uranium dioxide-molybdenum cermet, molybdenum encapsulated uranium dioxide, and uranium nitrides offer the opportunity to develop higher temperature reactors than zirconium-hydride reactors. Higher temperatures can be used to significantly lower power plant weight and volume. Core exit temperatures of 1675 K with lifetimes of 7-10 years are possible for the mid-1980's.
- Feasible concepts are available to eliminate possible single-point power plant failures. A potential major technology contributor would be the use of heat-pipes for transferring heat between parts of the power plant.
- Electrical converter technology has been advanced to the degree where 1275 K system temperatures are now being demonstrated for tens of thousands of hours. The isotope program has plans to develop thermoelectric converters of 10 to 15% by the mid-1980's. Also dynamic converters with efficiencies of as high as 35% are under development. A Brayton cycle without frictional surfaces has been run successfully for over 26,000 hours. By the 1990's, it is expected that converter technology will be advanced to the point where reactor exit temperatures of 1675 to 1875 K can be utilized.
- Extensive development of control drum actuators was performed on zirconium hydride-uranium hydride reactors. This work has continued to the present and is directly applicable to achieving a seven to ten year lifetime reactor control system.

- Shielding and reflector materials and design were highly developed in past space power programs (SNAP and ROVER). This demonstrated technology minimizes the amount of development effort required in future reactor power plants for space.

We believe development of reactor space power is well within current capabilities. A review of key components indicates that the material and design technologies have been tested in other programs. No single item was identified as a research item - one requiring development of entirely new technology. Though significant, the design problems all appear solvable. A brief assessment of the state of technology follows:

Reactor Core. Uranium-carbide fuels have been extensively developed for the nuclear rocket (ROVER) program. The work has continued for high temperature gas reactor (HTGR) fuel elements. The main problem foreseen is with fuel-cladding compatibility and in fabricating the fuel to the specific dimensions. Past experience has shown that control of impurities and atmosphere are important factors in establishing long-life fuel elements. Also, long life of fuel elements must be demonstrated.

The heat pipe was invented at LASL in 1963 and has had extensive use throughout the world, including space applications. Heat pipes will be mated with reactor fuel for the first time in this program, so good thermal contact between fuel and heat pipes must be demonstrated. There is a question of heat-pipe closure and proper control of impurities to ensure long life.

Reflector. The reflector will also use technology developed in the ROVER program.

Shield. Space-type shields have been developed in the ROVER and SNAP programs. Oak Ridge National Laboratory has performed extensive SNAP experimental work. The shield technology is considered to be mature.

Power Converter. ERDA and NASA have funded extensive work on power-conversion systems for the radioisotope, SNAP, and heat pipe/thermionic programs. The silicon-germanium thermoelectric technology appears to offer low risk for power levels to 50 kW(e). Dynamic systems such as the Brayton cycle can be used at higher power levels. Major development work on power converters would be continued with industrial organizations that have already demonstrated a high degree of competency in

the field. Thermoelectric modules at the desired power density for 7-10 year lifetimes must be demonstrated while using heat pipes. A Brayton converter has been run for over 26,000 hours. Thermionics are currently under development by NASA at the Jet Propulsion Laboratory.

Radiators. Radiator technology is pretty well developed but the extent of which heat-pipe radiators can be used must be assessed. Higher temperatures planned for radiators with thermoelectric converters will minimize their size and alleviate problems associated with folding or roll-up radiators. Brayton converters require lower temperature radiators in order to achieve high cycle efficiencies. As a result, Brayton converter radiators will probably be larger but use more conventional materials. Shield mass is the dominant power plant mass. Also, the shuttle can deliver large masses to low orbit. Therefore, advanced radiator technology is not an important need for the SCB power plants.

Controls. Drum actuators have been extensively developed by Atomics International and should satisfy our design requirements. Control sensors are within the state-of-the-art. Control electronics will be surrounded by thermal and radiation shields, so current technology can be used.

At present, conceptual design studies are being performed by LASL as a basis for selection of the next generation of space power reactors.

#### H. Probability of Success

A preliminary budget has been prepared for a space reactor program (Table VI). As stated in Section G, no items are placed in the research category but there are a number of difficult design problems. With adequate and timely funding, we have a high degree of confidence for success.

#### I. Change in Power Requirements

Minimum modifications will result from late changes in the power requirements. The reactor is being sized for the largest power level anticipated at 1000 kW(t) and could deliver 400 kW(e) if necessary at some future time without modifications. The shield could either be made thicker during manufacturing or moved further from the SCB to adjust for higher power levels. The converter could be replaced with either higher efficiency units or extra modules added. The radiator would need to be extended. If a foldable radiator is designed, this is a straightforward modification. The design of all components will consider future increases in demand.

TABLE VI  
SPACE REACTOR PROGRAM  
Very Preliminary Estimates

|                   | <u>1979</u> | <u>1980</u> | <u>1981</u> | <u>1982</u> | <u>1983</u> | <u>1984</u> |
|-------------------|-------------|-------------|-------------|-------------|-------------|-------------|
| System            | 5.4         | 16.6        | 18.6        | 22.4        | 22.8        | 25.7        |
| Conversion        | 2.2         | 5.0         | 6.5         | 7.5         | 7.6         | 8.5         |
| Safety            | <u>0.6</u>  | <u>1.2</u>  | <u>2.0</u>  | <u>2.2</u>  | <u>2.3</u>  | <u>2.5</u>  |
| Total Program     | 8.2         | 22.8        | 27.1        | 32.1        | 32.7        | 36.7        |
| Capital Equipment | 0.5         | 1.1         | 1.1         | 2.0         | 2.1         | 2.2         |
| Construction      | 1.0         | 3.5         | 3.0         | 2.0         | 1.5         |             |

Expected growth of power delivered can be achieved in a number of ways. The simplest, but most costly initially, is to launch the largest anticipated power module at the beginning of SCB. The power level of operation would then be varied by the control system. Such derating of operation would add additional conservatism to the plant operation and longer life can be expected at the lower power levels.

A second philosophy would be to add modules to the subassemblies as higher power levels are required. A possible growth pattern using this philosophy is discussed below.

For thermoelectrics, if one assumes an initial 50 kW(e) power module, there are a number of ways to increase the power level. These include:

- Adding radiator sections, thus reducing the cold junction of the thermoelectrics. Efficiency might be increased from 8% with a cold junction of 775 K to 12% with a cold junction of 475 K. This means that if the initial T/E design incorporates sufficient modules for 50 kW(c), that additional radiator sections can be added to obtain 75 kW(e) without changing the reactor shield or converter subassemblies.
- Replacing T/E modules with higher performance modules. Steady progress is expected in T/E materials and design. Therefore, after a period of time, one might expect increased power by replacing the T/E modules. Efficiencies of 15% are projected in the future.
- Replacing T/E modules with modules that have additional thermoelectric units. The reactor would operate at higher power so additional shielding must be provided as well as radiator sections. Power levels over 100 kW(e) can be achieved with early 1980's technology and 150 kW(e) with mid-1980's technology.

For the dynamic converter, growth can be achieved by:

- Adding radiator sections. If the system is initially designed to provide 50 kW(e) with 25% conversion and includes a radiator to provide a compressor inlet temperature of 450 K, a reduction of compressor inlet temperatures to 300 K would increase efficiency to 40%. Thus, a 50 kW(e) power module with added radiator sections would deliver 80 kW(e).
- Incorporating in the original reactor/dynamic heat exchangers the capacity to handle up to 1000 kW(t). Thus, a 25% converter unit could deliver 250 kW(e). Larger capacity converters would replace smaller units as needed.

- Adding an additional converter unit and radiator section. Provision would be necessary for an additional reactor heat exchanger in the original configuration along with shielding for higher power. The additional unit would provide 25% more power. At 25% efficiency, total power would be 75 kW(e) and at 40% efficiency, the power would be 120 kW(e).
- Replacing the converters with higher temperature units. The reactor can operate at higher temperatures but the current design limits are in the converter. Higher temperatures could increase efficiency to close to 45%. This means a 200 kW(t) reactor would provide 90 kW(e).

## J. Maintenance and Replacement

Two major philosophies were considered feasible for maintainability. The first is based on replacement of the power module except for the major biological-shield in case of a malfunction; the second would replace major subassemblies or components that are readily accessible and do not have high levels of induced radiation. Maintenance at the component level is considered undesirable in the near term because of the cost of setting up specialized facilities in space that would have little demand.

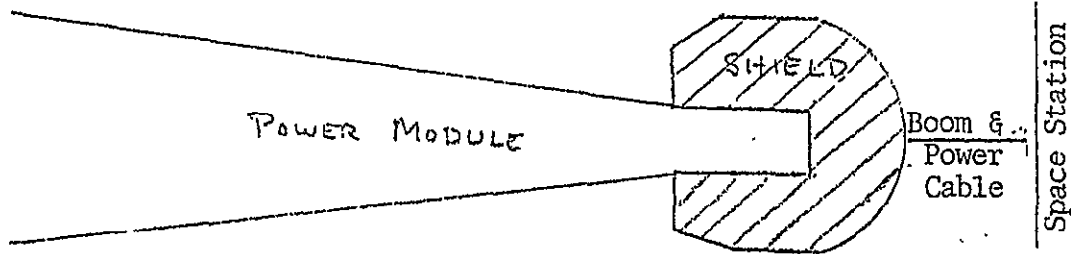
1. Replacement of Power Module. The power plant is designed for operation for the selected time period such as 10 years without maintenance. Because of the need for continuous power, dual power modules or an emergency power unit (each independent) will be needed in case a failure should occur at some time. With the expected high design reliability, failures should be separated by years of successful operation. When a failure does occur, the life of the remaining components and subassemblies will have been reduced from use and thus the mean time between failures in the future of the remaining subassemblies is less. Replacing the power module instead of a subassembly avoids lower mean times between failures. Also, new technology can be introduced into the power subassemblies and additional requirements met. The cost of a power subassembly is sufficiently low to justify replacing the module [\$4.5M for 50 kW(e)]. To replace a module would be relatively straightforward (see Fig. 17).

2. Major Subassembly Replacement. The reactor actuators, converters and sections of radiators are considered potential subassemblies that might be replaced. It is assumed that only limited access would be permitted in the immediate vicinity of these components and then only when the reactor is not operating and sufficient

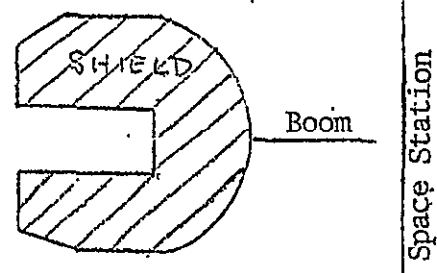
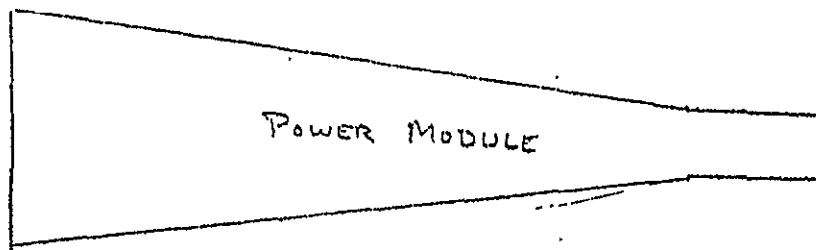


FIGURE 17

POWER-MODULE REPLACEMENT



1. Shutdown reactor and allow two days for fast decay product decay.
2. Manual lock on drums in shutdown position.
3. Disconnect cables to space station.
4. Attach removal rockets and disconnect power module lock from biological shield.
5. Move personnel to safest location in station such as directly in front of biological shield.
6. Activate rockets for perpendicular removal station to a safe distance.
7. Activate sun-disposal rockets.
8. Bring in new module using station crane.



POWER MODULE SEPARATED FROM SHIELD

time has passed to eliminate fast fission decay products (about two days). The amount of maintenance time would be limited to maybe 8 hours for a person to disconnect and replace a failed subassembly. Biological radiation standards would be strictly adhered to. Replacing major subassemblies has the advantage that the most probable failures will occur in the units and as such, a power plant could possibly give many more years of useful life. The major arguments against subassembly replacement are:

- Locating the exact failed unit may be difficult.
- Making provisions for replacing subassemblies may lead to a higher probability of failure. For instance, adding disconnects to the Brayton converter lines may introduce a major possible failure point. Experience shows that line connectors, and valves are major sources of failures.
- The cost of developing special disconnect and connection equipment may exceed the cost of replacing the complete power module. The number of times one anticipates performing these operations is small. For instance, if each Brayton converter has a reliability of 0.95 with a lifetime of 7 years, then each unit has an expected mean time between failure of 136 years. If three independent converter units are provided, it becomes questionable whether one wants to develop specialized equipment to replace subassemblies in space.

#### K. Reliability and Lifetime

The power system is currently being designed for seven years life with a reliability of 0.95. The system can be designed for a different specification if desired. The major subassemblies are being designed as follows:

|                         |             |
|-------------------------|-------------|
| reactor core and shield | 0.98        |
| converter               | 0.99        |
| radiator                | 0.99        |
| power regulator         | <u>0.99</u> |
| Overall                 | 0.95        |

Fuel burnup after seven years at 1000 kW(t) is 4.3%. The lifetime of the core (without failures) based on fuel burnup is 35 years for 50 kW(e) and 12 years for 150 kW(e).

The power module is being designed for end-of-life conditions. This means that the design already considers that a number of heat-pipes in the core might fail, the radiator will lose a certain number of stringers from meteorites, etc. The power of

the system is regulated during its lifetime by varying the amount of energy generated in the core. Thus, the beginning and final electric power output are programmed to be the same.

Emergency cooldown systems are unnecessary because of the redundancy built into the unit. The design is aimed at eliminating single point failures.

#### L. Thermal Power for SCB

Thermal power could be provided at converter inlet temperatures. A trade-off would be necessary between pumping the thermal power to the SCB and the shielding mass.

#### M. Safety

1. Power Plant. Redundancy is the key to providing hardware systems with good safety characteristics. For space systems, however, the number of levels of redundancy must be balanced against resulting increases in weight and complexity, which themselves will reduce the probability of mission completion. Safety features proposed for the nuclear system to prevent a reactor criticality accident are as follows:

- The control drums are to be pinned in the shutdown position with captive-key locks for all noncritical earthbound nuclear operations. It should be possible to unlock and operate one drum at a time for testing. The pins would be removed before launch.
- The AI control-drum actuator has a brake that holds the drum in position until the brake is energized. The motor does not have enough torque to drive the drum with the brake engaged.
- With six control drums, shutdown can be satisfactory with some of the drums inoperative.
- A separate shutdown controller will be provided to ramp the control drums to their shutdown position if the normal control system fails. This controller should not share electronic components with the normal controller and should have an alternative energy source. The shutdown controller should be able to override any actions of the normal controller after an emergency shutdown is initiated. The system should be capable of being reset to normal operation after an emergency shutdown.

2. Ground and Launch Operations. Safety during ground and launch operations is a problem to consider; however, safety problems have been solved as demonstrated in the flight test of the SNAP-10A reactor. The safety program will closely parallel that described for isotopes with test to show that launch pad fires, propellant fires, impact capability at terminal velocity during a launch abort, and water impact can all be safely handled. The reactor will not be run for any significant period prior to launch. Thus, it will not achieve appreciable radioactivity before flight and can be safely handled on the ground. The reactor will not have developed appreciable fission products in case of a shuttle malfunction. The uranium fuel poses a minimal safety hazard.

3. Orbital Operations. Safety is provided during operation by providing redundancy throughout the system. For instance, the core would include many heat pipes (91) arranged so that the failure of single heat pipes would result in the surrounding heat pipes removing the thermal energy. Provision is made in the design such that a melt down of the core cannot occur - there is no need for a separate emergency cooldown system. With separate emergency power source or a duplicate reactor power module on-board the SCB, there should be no need to operate the power module with sufficient components failed such that another failure would lead to an unsafe condition. For instance, if three Brayton converters are included in each power module, once two have failed the system would be shutdown for replacement or maintenance rather than run the risk of losing the coolant of the last converter. The reactor power industry is the safest in the country and it is expected that the high standard established would be applied to any space program.

#### N. Fluid Freezing

The solidification of the fluids in the heat pipes are not a problem. This has been demonstrated many times.

#### O. Controls and Instrumentation

1. Design Considerations. The current concept is to control the reactor with six rotating drums located in the reflector. Each drum has an arc segment of neutron-absorbing material on one side. Reactivity is increased or decreased by rotating the neutron-absorbing material away from or toward the core, respectively. Each drum will be rotated with a stepping-motor actuator located on the side of the shield for protection from the high nuclear and thermal radiation. The maximum positive reactivity-rate capability of the control actuators will be very low for safety reasons. On the basis of ROVER experiments, this approach looks very feasible.

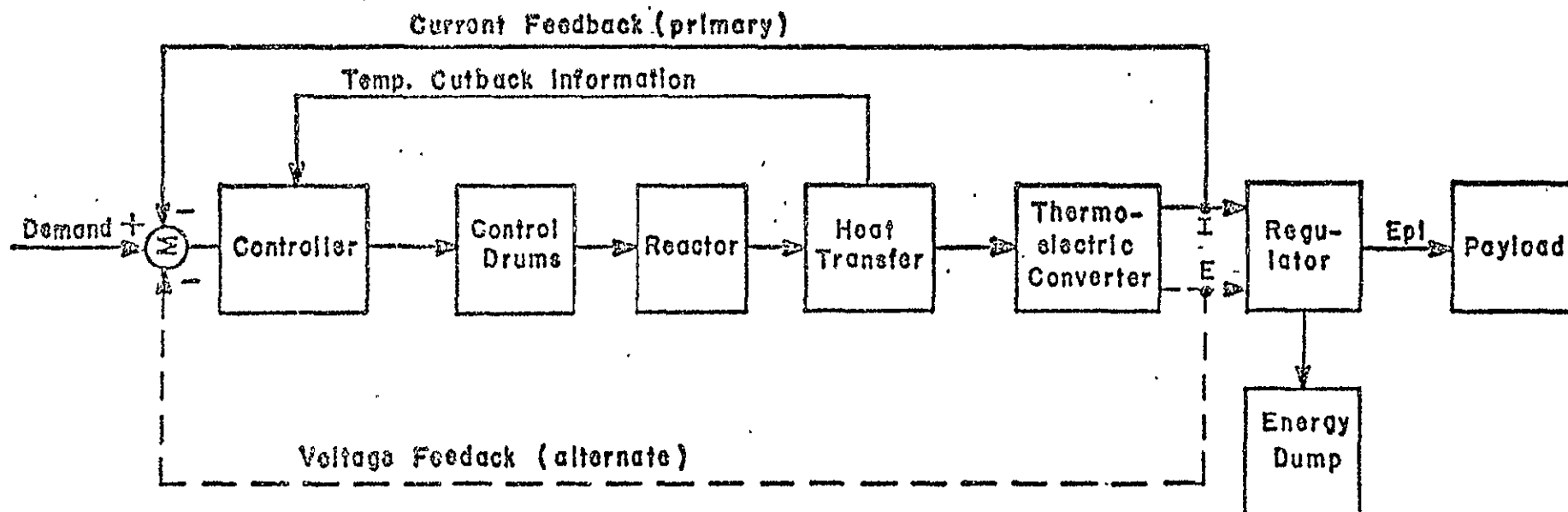
To provide guidance in the controls area, we assumed the ultimate output requirement would be a payload voltage of 28 V for thermoelectrics and 120 V dc for Brayton converters regulated over a power range from full power down to some nominal lower value, possibly zero. Two control approaches considered for the system are described below:

1. Operate the control drums under closed-loop control to produce a constant output current. The payload voltage would be regulated by dumping energy not needed by the payload into a payload heat radiator. This function would be handled with a high-performance series or shunt regulator located in the payload module. A temperature-cutback capability would be provided to prevent overheating of high-temperature systems. This approach has the advantage that high-temperature systems operate at essentially constant temperatures. It requires that a radiator be able to radiate the full-thermal energy generated by the reactor, but this capability would probably be needed anyway.

2. Operate the control drums as required to provide the correct voltage to the payload with a slow control loop. A shunt regulator would be used to provide high-speed regulation of payload voltage. The steady-state power dissipated in the shunt regulator would vary from zero for full payload power to several kilowatts at reduced power. As with system 1, a temperature cutback capability will be necessary. This control approach has the advantages of (1) operating the high-temperature systems at lower temperatures, (2) operating the thermoelectric elements at lower currents, and (3) reducing the payload radiator heat load. It has the disadvantage of producing temperature changes, at slow rates, in the high-temperature systems.

Information now available is insufficient for proper evaluation of the two approaches. Both approaches will require temperature control based on thermocouples in or near the reactor core for startup. Figure 18 is a simplified block diagram of the control system.

The control and instrumentation electronics will mostly be digital circuits located in the space station. The system will probably operate under the control of a microprocessor for normal operation. A shutdown controller will be provided to scram the control drums if an emergency shutdown situation is detected.



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FIGURE 18  
PRELIMINARY BLOCK DIAGRAM OF CONTROL SYSTEM

2. Control Actuators. The stepping-motor-actuator development work done by Atomics International (AI) for the SNAP program has established a good technology base for the control actuators. Continued low-level funding of this work has enabled AI to accumulate a lot of test time on their actuator. This unit is a stepping motor with an output torque of 0.71 N-m (100 oz-in.) and a step size of 31.41 mrad (1.8 degrees). It requires an excitation of 12 V and 5.5 A, and it weighs approximately 3.5 kg (8 lb). It has an integral brake and the first stage of gearing. Most of the AI high-temperature testing has been done at 810 K, but AI has also done considerable testing at 975 K with no failures.

We believe the design approach for the control-drum actuator should be to use the AI actuator in its present form. Heat sinking the actuators to the 875 K converter radiator will probably afford sufficient cooling. If thermal analysis indicates otherwise, a separate heat-transfer and radiating system will have to be provided for the actuators.

3. Instrumentation. Control information for vehicles in space will be derived from thermocouples on the power plant and from voltage and current measurements in the space station. As voltage and current measurements will be made with conventional instrumentation, they will be discussed no further.

The primary parameter for the temperature control is the temperature of the thermoelectric-converter package. This temperature should be around 1275 K. The alternative candidate parameter is the core fuel temperature, which is predicted to be 1390 K. The control temperature will be the average of probably 10 to 20 measurements to increase the validity of the measurement and improve the long-term reliability. Thermocouples that deviate excessively from the average will be rejected from the averaging calculation.

Temperature measurements using thermocouples are well within the state-of-the-art. The major problem will be to determine, by information search or by testing, if long-term stability problems will require the use of refractory-material thermocouples rather than inexpensive medium-temperature devices. The primary transducer candidate is the Chromel-Alumel thermocouple with an Inconel-600 sheath and high-density MgO insulation. The lowest melting temperatures in this system are 1630 K for Inconel 600 and 1620 K for Chromel. It may be desirable to have a backup thermocouple with refractory materials.

The shaft positions of the control drums will not be required as control parameters, but it would be highly desirable to have them for startup and safety reasons. Several transducer approaches are feasible.

Instrumentation will be extensive to satisfy safety and diagnostics needs. Some of the measurements required will be neutron flux, core temperature distribution, shield temperatures, heat-pipe differential temperatures, actuator temperatures, reflector temperatures, and perhaps some strain measurements.

We expect the electronics associated with the control and instrumentation systems to be located in the payload module. The payload radiation level is low enough to permit the use of conventional electronic components. Redundant electronic subsystems will be used where necessary to avoid what would otherwise be catastrophic effects of single failures.

4. Startup and Shutdown. The control system will be capable of multiple restarts both on the ground and in space.

The startup scheme for vehicles in space will evolve from analysis of the problem and from information obtained during ground tests. The problem will be complicated by the possibility that the heat-pipe fluid will be frozen and by the cold critical control drum-position uncertainty caused by fuel burnup and fission-product poisoning. The time allowed for startup will determine the type of system to be developed. As an example, one possible scheme is described below:

- Ramp the control drums relatively fast from shutdown to a position well below cold critical.
- Ramp the drums through cold critical very slowly.
- When the temperatures at the core end of the heat pipes reach the minimum controllable value, switch to temperature control.
- Raise the control temperature enough to cause thermal activation of the heat pipes by conduction within a reasonable time.
- When differential temperatures along the heat pipes indicate the pipes are operating, increase the control temperature enough to permit switching to the high-power control parameter.

Several good options are available for shutting down under nonemergency conditions. Two obvious methods are simply to ramp the control demand to zero or to ramp the control drums to the shutdown position. The ramps would be sufficiently slow to avoid excessive thermally induced stresses.



## Appendix E

### COST ESTIMATING GUIDELINES

The task costs documented in this study report are from two major sources--LASL and LeRC. These agencies supplied the cost data for the basic reactor brayton and solar brayton units. Cell efficiencies and cost per watt of power generated by the low-cost solar cells were specified by NASA to be \$5 per watt and \$50 per watt, respectively. These were incorporated as furnished. The costs for the supporting structure and equipment were developed by MDAC.

The costs for solar cells and arrays were based on data obtained from Lockheed. The cost of batteries was derived from data supplied by Eagle Picher. Estimates for fuel cell costs were supplied by General Electric and Pratt & Whitney.

MDAC used information from its data bank to calculate the other costs required for the systems. These included costs for structure, mechanisms such as gimbals and joints, environmental/thermal control equipment, electronics, power conditioning and distribution, and attitude control subsystems. These costs were usually derived from the Phase B Space Station Studies and the recently completed Space Station Systems Analysis Study.

These cost estimates are based on general descriptions of the hardware and indicate the order of magnitude for the various items. More emphasis was placed on predicting the comparative costs of the various systems than on defining the absolute costs. In implementing this approach, uniform factors were applied across all items to convert from estimated manufacturing costs to total program cost. More detailed technical and cost investigation would be required to define which items require more testing and which require less, which require more integration and which less, etc.

The solar array and support structure includes the solar cells, the blankets, and the conductors connecting the cells. The supporting structure includes only the structure required to hold the array and, if it is deployable, the masts and extending mechanisms and controls.

The energy storage and power conditioning item includes the batteries, fuel cells, electrolysis units, chargers, regulators, inverters, buses, and wiring associated with distributing and controlling the power generated by the solar panels and the power sources used while the solar arrays are in earth shadow.

Structure/mechanical/miscellaneous costs cover all hardware costs not included in the solar array and energy storage items. These include general structure and mechanisms such as docking ports, gimbals, enclosed areas (e.g., access tunnels and compartments), and equipment canisters. Also included are any environmental control equipment, thermal coatings, or meteoroid shields incorporated in the system.

The free-flight support item costs include all the controls, sensors, telemetry, and reaction and attitude control equipment required to permit the system to be left in orbit without being connected to the Shuttle.

The spares and replacement hardware are calculated for a 10-year life. This assumes the batteries have a 3-1/3-year life and the fuel cells a 5-year life. The spares are assumed to be 10 percent of the total cost of the 3 sets of batteries, 20 percent of the total cost of the 2 sets of fuel cells and electrolysis units, and 25 percent of the costs of all other electrical items. (This means the spares factor is 30 percent of the cost of a flight set of batteries, 40 percent of the cost of the fuel cells, and 25 percent of the cost of other items.)

The charts comparing different battery types reflect the different costs associated with the development of the storage units and the power conditioning equipment required for them. The change in cost for the  $\text{NiH}_2$  batteries reflects a decrease in cost for developing the simpler charges (offset by increased development costs for the batteries themselves). The development cost for the fuel cell system reflects an increase in the tankage, and a decrease due to the elimination of the battery chargers and the difference in the development cost of the  $\text{NiCd}$  batteries and the fuel cells. The development cost of the solar array is assumed to be independent of the type of power storage used. A minor variation probably exists between the different systems, but it was not identified in this study.

The production cost of the array varies with the type of power storage. It reflects the change in the array area associated with the various battery/fuel cell performance. The spares/replacement cost reflects the result of the changes in both the battery/fuel cell and power conditioning costs.

The change in the cost of the deployable array system with the type of solar cell used reflects the cost objective specified by NASA. It was assumed that the development cost for the array would not vary significantly for the

different types of cells. The production cost of the flight unit reflects the cost change associated with the change in size as well as the efficiency of the cells.

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